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NAVAER 01-60JKE-502

6 86

Handbook
Maintenance Instructions
NAVY MODEL
FJ-4B
AIRCRAFT

SECTION VI
INSTRUMENTS
AND
RELATED SYSTEMS

PUBLISHED BY DIRECTION OF
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1 August 1957
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Page No.	Date of Latest Revision
ii	1 August 1958
*iii	1 November 1958
*iv	1 November 1958
*v	1 November 1958
6-6	1 May 1958
6-6A	1 May 1958
6-6B	1 May 1958
6-6C	1 May 1958
6-6D	1 May 1958
6-7	1 August 1958
6-8	1 August 1958
6-9	1 August 1958
6-10	1 August 1958
6-11	1 August 1958
6-13	1 August 1958
6-14	1 August 1958
6-19	1 August 1958
6-20	1 August 1958
6-21	1 August 1958
6-25	1 February 1958
6-28	1 November 1957
6-31	1 May 1958
6-32	1 August 1958
6-33	1 August 1958
6-34	1 August 1958
6-36	1 August 1958
6-37	1 May 1958
6-38	1 May 1958
6-39	1 May 1958
6-40	1 August 1958
6-40A	1 May 1958
6-40B	1 February 1958
6-42	1 May 1958
6-42A	1 May 1958
6-42B	1 May 1958
6-43	1 May 1958
6-44	1 May 1958
6-46	1 February 1958
6-52	1 August 1958
*6-52A	1 November 1958
6-52B	1 November 1957
6-53	1 November 1957
6-54	1 May 1958
*6-55	1 November 1958
*6-56	1 November 1958
*6-56A	1 November 1958
*6-56B	1 November 1958
*6-57	1 November 1958
6-58	1 May 1958
6-62	1 May 1958
6-63	1 May 1958
6-69	1 November 1957
6-70	1 November 1957
6-71	1 November 1957
6-72	1 November 1957
6-73	1 May 1958
6-89	1 May 1958
6-90	1 May 1958
6-91	1 May 1958
6-92	1 August 1958
6-93	1 May 1958
6-94	1 August 1958
6-94A	1 August 1958
6-94B	1 August 1958
*6-95	1 November 1958
6-96	1 May 1958
6-97	1 May 1958
6-98	1 May 1958
6-99	1 May 1958
6-100	1 May 1958
6-100A	1 May 1958
6-100B	1 May 1958
6-101	1 May 1958
6-102	1 August 1958
6-103	1 May 1958
6-104	1 August 1958
6-104A	1 May 1958
6-104B	1 May 1958
6-105	1 May 1958
6-106	1 August 1958
6-107	1 May 1958
6-108	1 May 1958
6-109	1 May 1958
6-110	1 May 1958
6-110A	1 May 1958
6-110B	1 May 1958
6-110C	1 May 1958
6-110D	1 May 1958
6-110E	1 May 1958
6-110F	1 May 1958
6-110G	1 May 1958
6-110H	1 May 1958
6-110J	1 May 1958
6-110K	1 May 1958
6-111	1 May 1958
6-112	1 May 1958
6-113	1 May 1958
6-114	1 August 1958
*6-114A	1 November 1958
6-114B	1 May 1958
6-114C	1 May 1958
*6-114D	1 November 1958
6-115	1 November 1957
6-117	1 November 1957
6-127	1 May 1958
6-128	1 May 1958
6-129	1 November 1957
6-130	1 November 1957
6-132A	1 May 1958
6-132B	1 May 1958
6-133	1 November 1957
6-134	1 November 1957
6-138	1 August 1958
6-139	1 August 1958
6-140	1 August 1958
6-141	1 August 1958
6-142	1 August 1958
6-143	1 August 1958
6-144	1 August 1958
6-145	1 November 1957
6-147	1 November 1957
6-152	1 February 1958
6-152A	1 February 1958
6-152B	1 February 1958
6-158	1 August 1958
6-165	1 February 1958
6-166	1 February 1958
6-166A	1 February 1958
6-166B	1 February 1958
6-168	1 August 1958
6-170	1 November 1957
6-171	1 November 1957
6-172	1 November 1957
6-175	1 August 1958
6-176	1 August 1958
6-177	1 November 1957
6-178	1 August 1958
6-183	1 February 1958
*6-184	1 November 1958
6-184A	1 February 1958
6-184B	1 February 1958
6-186	1 May 1958
*6-187	1 November 1958
*Index 1	1 November 1958
*Index 2	1 November 1958
*Index 3	1 November 1958
Index 4	1 August 1958

*The asterisk indicates pages revised, added or deleted by the current revision.

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Introduction to SECTION VI

THIS HANDBOOK IS ONE OF A SERIES OF TEN which contain instructions required by using activities for the maintenance of Model FJ-4B aircraft. These are systems type handbooks. Each system in the aircraft is covered completely in a particular handbook. This includes all hydraulic, pneumatic, mechanical and electrical portions of the system. This has been done in order to assist the mechanic in becoming familiar with and in maintaining all phases of each system.

The "Instruments and Related Systems" handbook contains only nonclassified instrument maintenance data which is peculiar to FJ-4B aircraft. For maintenance data on related electronics systems, refer to Section IX of the FJ-4B Handbook of Maintenance Instructions.

This handbook contains information necessary for the performance of class C and class D maintenance on those items of Contractor Furnished Equipment for which there are no separate handbooks. This handbook does not contain instructions for the overhaul of components. Such instructions are contained in separate handbooks of overhaul instructions for the individual equipment.

Instructions for the repair of aircraft structure are contained in the Handbook of Structural Repair (NAVAER 01-60JKD-503) for these aircraft.

Data necessary for obtaining replacement parts and complete identification of parts are contained in the Illustrated Parts Breakdown (NAVAER 01-60JKD-504) for these aircraft.

Weight and Balance Data are found in the applicable AN 01-1B-40 handbook for each of these aircraft.

To identify and obtain these publications and handbooks covering separate items of equipment, refer to the Naval Aeronautic Publications Numerical Index (NAVAER 00-500).

BuAer Serial Numbers 139531 through 139555, 141444 through 141489 and 143493 through 143643 have been assigned to the FJ-4B. In addition, a lower case letter has been made a part of each serial number as it is painted on the aircraft. These lower case letters have been assigned to blocks of serial numbers as follows:

SERIAL NUMBER	LETTER
139531 through 139555	i
141444 through 141489	j
143493 through 143542	k
143543 through 143593	l
143594 through 143643	m

TABLE OF CONTENTS

Section VI INSTRUMENTS AND RELATED SYSTEMS (Cont)

Section VI INSTRUMENTS AND RELATED SYSTEMS (Cont)

Section VI INSTRUMENTS AND RELATED SYSTEMS (Cont)

FJ-4B Airplane

GENERAL INFORMATION

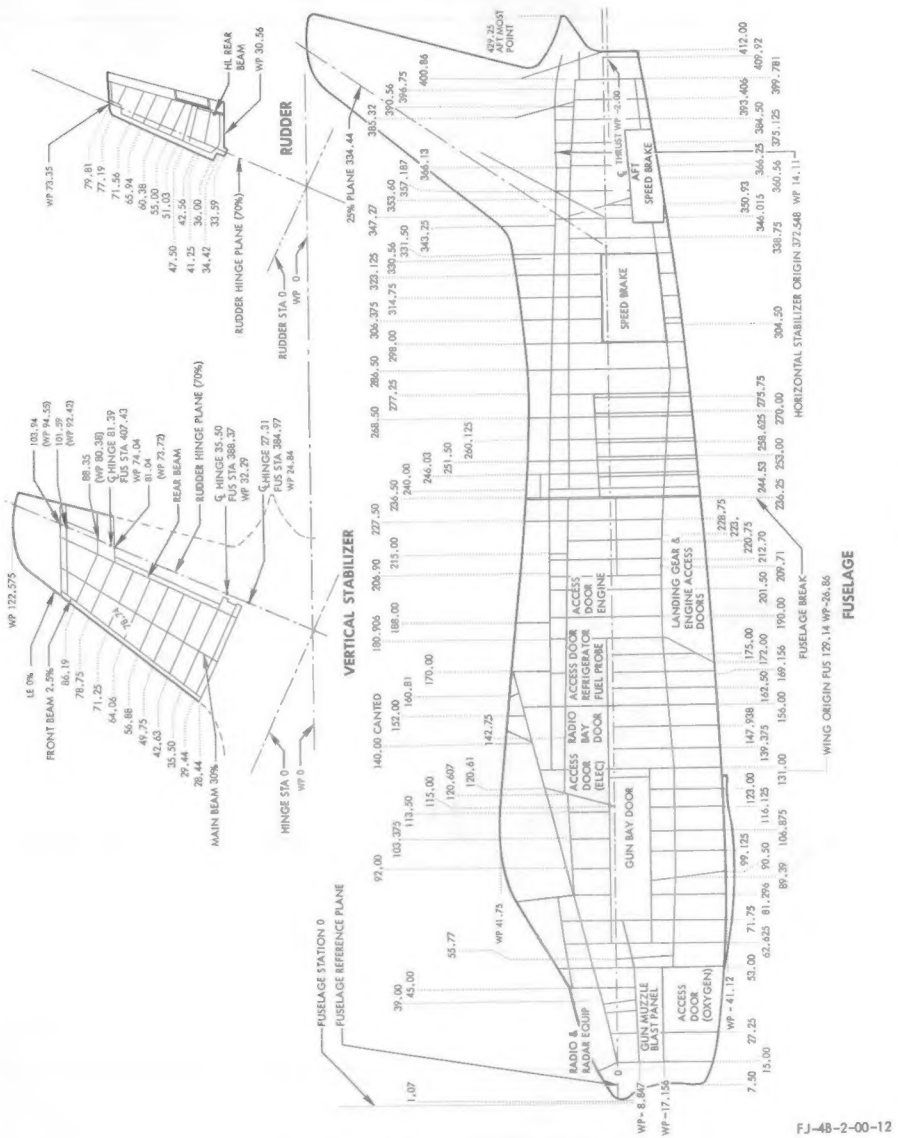

Figure No. 6-1. Airplane Stations (Sheet 1)

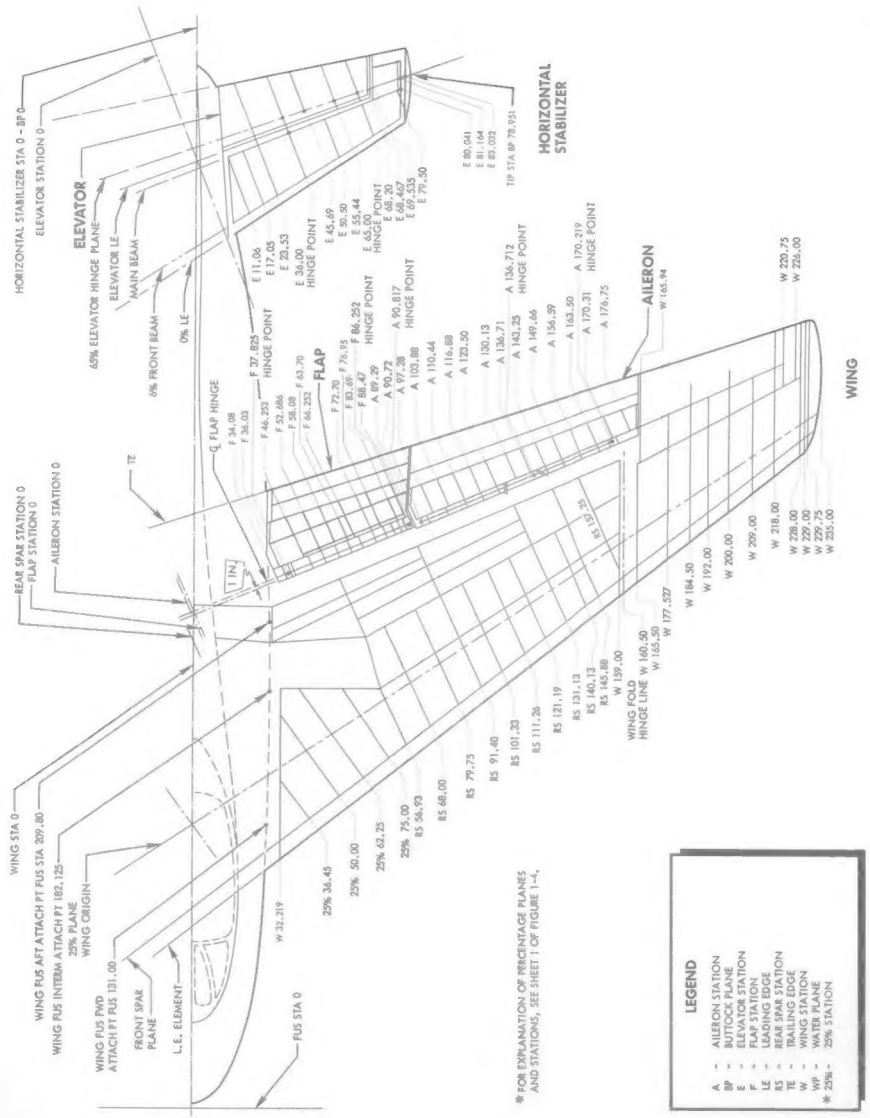

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Figure No. 6-1. Airplane Stations (Sheet 2)

Warning Ground safety locks and pins are to be installed at all times, except for flight and gear retraction check. Remove immediately before flight and stow in cockpit map case.

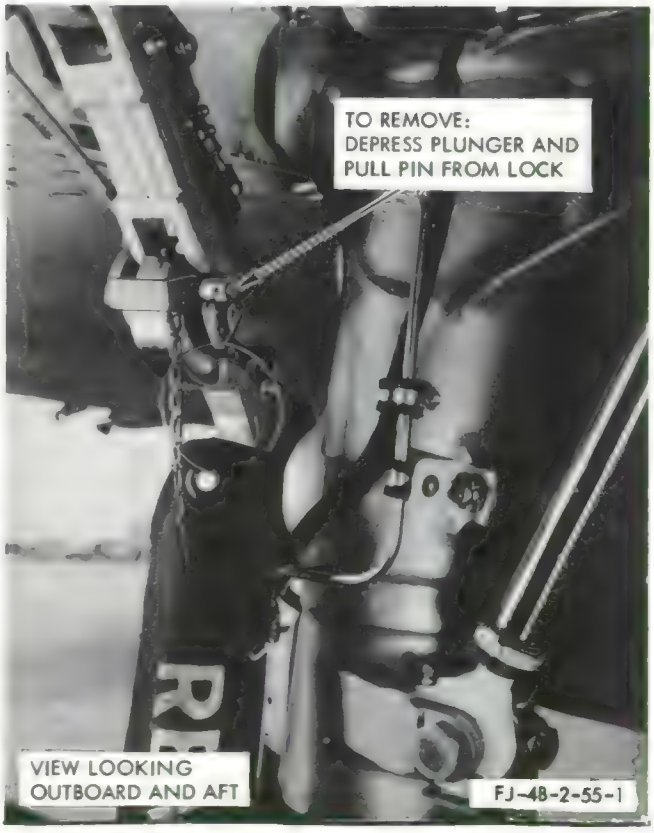
A time-saving method for performing certain testing procedures on the airplane (which normally would require the use of ground jacks) may be accomplished by disabling the ground safety switch. Attach a red warning flag, similar to the flags used on the landing gear ground safety locks, whenever the ground safety switch is disabled.

Warning When a red warning flag has been attached to the ground safety switch to indicate a disabled switch, never remove flag from the unit until switch has been properly connected.

GROUND SAFETY SWITCH

NOSE LANDING GEAR GROUND SAFETY LOCK

MAIN LANDING GEAR GROUND SAFETY LOCK



Note There is no ground safety lock for the arresting gear.

Figure No. 6-2. External Ground Safety Locks and Pins

Warning

- Keep out of the cockpit unless maintenance is required.
- Always consider the emergency escape system loaded and armed.
- Know where the safety pins are and be certain of their installation.
- Do not manipulate linkage wihout full knowledge of the emergency escape system.
- Do not use linkage or handles as handgrips.
- The catapult cartridge, canopy remover, remover initiators and exactor are ordnance items and should be checked and maintained only by qualified personnel.

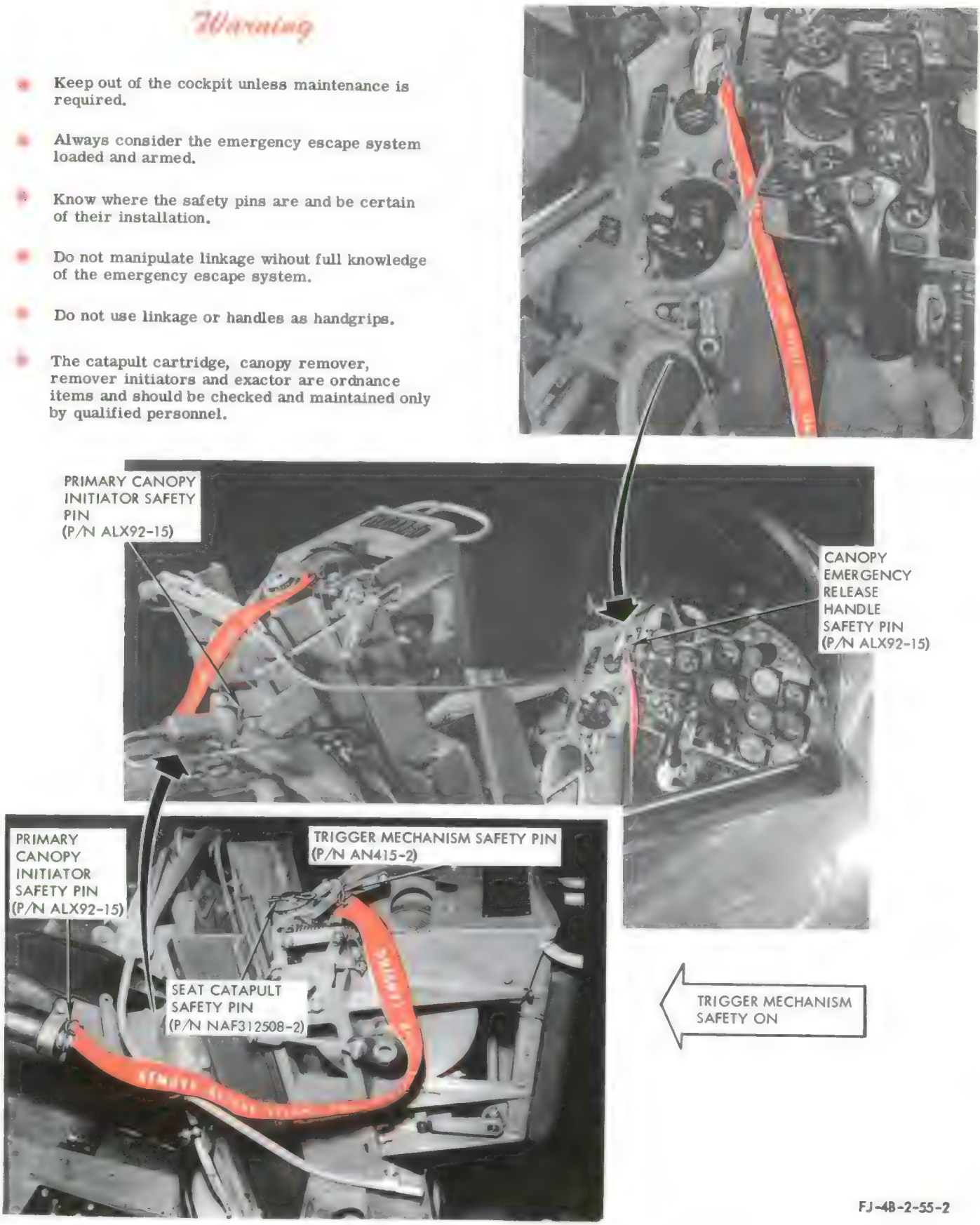
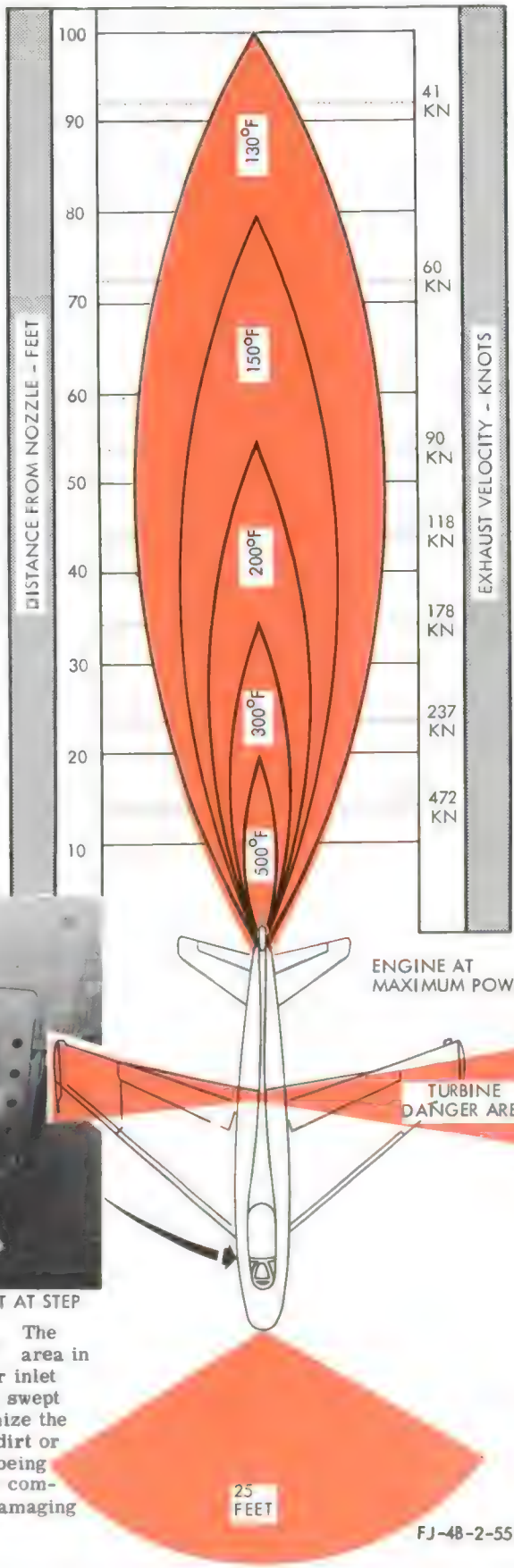


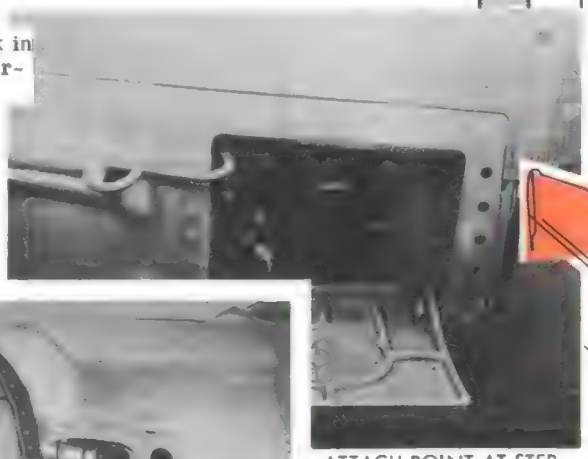
Figure No. 6-3. Emergency Escape System Ground Safety Pins

Warning

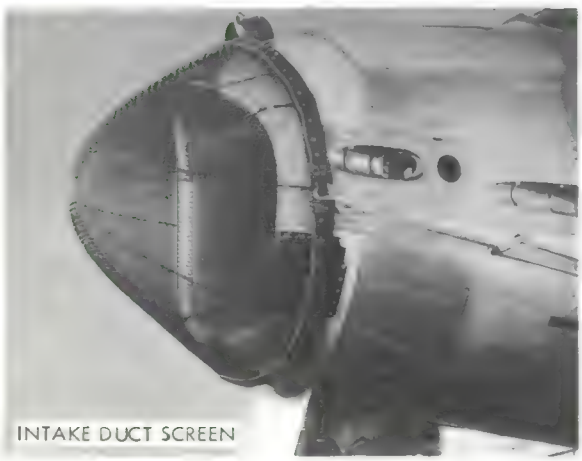
- Do not stand near the front of the air inlet duct while the engine is operating.
- Always approach the airplane from the side but not in the plane of rotation of the turbine when the engine is running.
- Avoid wearing hats or other loose clothing when working in the run-up area.
- Do not carry loose articles such as pencils, key rings or tools when near the air inlet duct.
- Do not foolishly experiment with the margin of safety by standing near, or feeling with your hand, the suction created by the engine.
- Do not stand on wing of the airplane while engine is operating, unless assistance is required during cockpit check-out or functional check of equipment.
- The loudest sustained noise produced by man is the noise of a jet engine operating at high rpm. Jet-engine noise is dangerous to personnel working in the immediate area. At distances from 50 to 200 feet, wear ear plugs and at distances within a radius of 50 feet, wear ear plugs and a type of over-the-ear protector. Prolonged exposure to jet-engine noise can cause pain and damage to the inner ear. Other effects of prolonged exposure are fatigue, nervousness and impairment of hearing.
- Do not stand at the edge of the blast area as the temperature could suddenly increase with engine speeds.



Place retaining rope hook in existing hole located in forward frame of step.



ATTACH POINT AT STEP



INTAKE DUCT SCREEN

Caution The area in front of the air inlet duct should be swept clean to minimize the possibility of dirt or other objects being drawn into the compressor and damaging the engine.

Figure No. 6-4. Ground Run-up Danger Areas

TEST POINT TROUBLE SHOOTING.

To ease and expedite electrical maintenance, test point trouble shooting data has been incorporated in system trouble isolation procedures and system wiring diagrams. As any system failure or malfunction may result from any one or a combination of electrical, hydraulic, pneumatic or mechanical reasons, all probable causes (reasons) for a stated trouble are covered in the same trouble isolation chart. There are three types of test points: major, secondary and minor. Textual references to these test points are made within each system trouble shooting paragraph and the specific location of each test point may be determined by referring to the appropriate system wiring diagram in Section X. No test point designation will be duplicated nor will more than one test point designation be given to any test point.

MAJOR TEST POINTS.

Major test points are used to isolate a power system failure to a physical portion of the airplane or to a group of systems. Major test points are symbolized on system wiring diagrams by a star encircled Arabic numeral. Major test points are referred to in text as: test point 1, test point 2, etc. Some examples of major test points are: generator and inverter outputs, power distribution connections, etc.

SECONDARY TEST POINTS.

Secondary test points are used to isolate failure to a specific system or to a specific item within a system. Secondary test points are symbolized on system wiring diagrams by an encircled capital letter(s). The letters "I" and "O" are not used to avoid confusion with the numerals one and zero. Secondary test points are referred to in text as: test point A, test point AB, etc. Some examples of secondary test points are: power inputs to individual units, tie-ins with parallel or interrelated systems, sequence switches, etc. Secondary test points for any specific system will always have their initial identifying letter the same letter as the initial letter of the wire numbers of that system.

MINOR TEST POINTS.

Minor test points are used to isolate failure within a unit. Minor test points are symbolized on system wiring diagrams by an encircled capital letter and Arabic numeral. The letters "I" and "O" are not used to avoid confusion with the numerals one and zero. Minor test points are referred to in text as: test point A1, test point A2, etc. Some examples of minor test points are: continuity through a switch or a relay that is part of a unit, resistance readings of items within a unit, etc. Minor test points for any specific system will always have as their initial identifying letter the same letter as the initial letter of the wire numbers of that system.

USE OF TROUBLE SHOOTING CHARTS.

The best trouble shooting aid is preventive maintenance and cleanliness. The next best trouble shooting aid is thorough knowledge of the theory and operation of the system in question. A thorough knowledge of the system permits rapid determination of the most likely probable cause for any given trouble and thereby reduces trouble shooting time and effort. The third most important aid is safety; observe all safety rules, check to make sure that the airplane and any attached ground power equipment is properly grounded, check to make sure that all ground safeties are installed, follow the trouble shooting instructions and if it is a two-man job, get another man to help. What is the trouble? Check the squawks, observe or perform an operational or functional check of the system in question. Check the trouble shooting charts of the system for the determined trouble. Select the most probable cause(s) and proceed to isolate the trouble; set up the system as specified in the "System Conditions" portion of the chart. Use the appropriate meters. Do not make ohmmeter tests or continuity checks on an electrically "hot" airplane. Complete check-out of the system in question without correction of the trouble may indicate that a parallel or interrelated system is at fault. If so, refer to that system for appropriate trouble shooting information. When a remedy is performed that does not correct the trouble, select the next most probable cause and continue trouble shooting. Isolation procedures are set up to require a minimum of effort. Each procedure should either isolate the trouble itself or isolate the portion of the circuit that contains the trouble. When a test point procedure is called out for an item (for example, a valve solenoid), parts of that procedure not spelled out which may lead to isolating the fault are: visual inspection for signs of physical damage, check of the ground connection or bonding and a check for good electrical connections. Similarly, when test points are called out for relay terminals, the switch section of the relay involved should be checked for proper action and continuity. The various portions of the trouble shooting charts and their functions are as follows:

a. TEST EQUIPMENT. This portion of the charts contains a list of all test equipment that will be required to perform any isolation procedure that follows on the same chart.

b. SYSTEM CONDITIONS. This portion of the charts specifies the desired system conditions for the tests that will follow. Some isolation procedures may require a change to these conditions; if so, the new conditions will be given in note form.

c. TROUBLE. This is the observed symptom, malfunction, or fault.

d. PROBABLE CAUSE. The probable cause(s) states the condition or reason causing the trouble. Probable causes are listed in their most likely order. The probable causes may be electrical, mechanical, hydraulic, pneumatic, etc, or a combination of these reasons.

e. ISOLATION PROCEDURE. This portion of the charts is a positive statement of action. If the probable cause is nonelectrical, there will be no mention of test points; if electrical, specific directions related to one or more test points will be given. Isolation procedures are listed in their most likely or accessible order. What meter is to be used will be determined by the required

meter reading(s). Use the appropriate system wiring diagram in Section X to locate test points and to perform wire segment continuity checks. Many isolation procedures require the use of test points located at a connector. In such cases, it is necessary to disengage the connector and to apply the test probe to the plug or receptacle portion of the connector as shown on the system wiring diagram. Connectors should never be disengaged with electrical power applied to the airplane. Do not damage connector sockets by inserting test probes.

f. METER READING. If the isolation procedure is nonelectrical, this portion of the chart will indicate that none is required. If test points have been specified in the isolation procedure, the value and type of reading will be stated. Resistance and voltage readings are the type most commonly required for the isolation procedures; values given will indicate their type and the corresponding type of meter should be used to obtain the reading.

g. REMEDY. For nonelectrical isolation procedures, the remedy will indicate the maintenance action required depending upon the results of the isolation procedure. For electrical isolation procedures, the remedy will indicate the maintenance action required for the meter reading obtained. Most remedies will indicate a definite maintenance action, but some remedies will indicate that further isolation procedures should be performed. Some meter readings will indicate that the airplane wiring is at fault (open or shorted) and the remedy will be to perform a wire segment continuity check. Such continuity checks should be performed so as to minimize effort. Remove power and disconnect wires as necessary; then, check for continuity at the most accessible mid-point of the circuit; in this manner, several wire segments can be checked for continuity at one time.

WARNING

Never disconnect wires or disengage disconnects with electrical power applied to the airplane. Always ground the airplane and any attached ground power equipment.

Note

Secondary test points are listed alphabetically and opposite to each applicable wiring diagram title. Figure numbers of the wiring diagrams listed can be found in the Wiring Diagram Index of Section X of this handbook. Major test points, not listed, can be found in the Starting and D-C Generating System, the D-C Power Distribution System and the A-C Power Supply and Distribution System wiring diagrams. Minor test points, also not listed, can be found by associating them with similar secondary test points.

TEST POINT	WIRING DIAGRAM TITLE
ALA-ALZ	Exhaust Temperature Indicating System
CPA-CPZ	Rudder Pedal Shaker System
D, DA-DZ	Hydraulic Pressure Indicating System
DLA-DLZ	Oxygen System
EFA-EFZ	Fuel Flow Indicating System
EPA-EPZ	Oil Pressure Indicating System
EQA-EQZ	Fuel Quantity Indicating System
ETA-ETZ	Engine Tachometer Indicator
F, FA-FZ	Vertical Gyro System
FAA-FAZ	Angle-of-Attack and Angle-of-Sideslip Indicating System*
FCA-FCZ	Polar Path Compass System
FLA-FLZ	Approach Light System*
FMA-FMZ	Angle-of-Attack and Angle-of-Sideslip Indicating System†
FNA-FNZ	Approach Light System†
FPA-FPZ	Pitot Heater
PBA-PBZ	D-C Power Distribution System (Right-hand forward console)
PDA-PDZ	D-C Power Distribution System (Left-hand forward console)
PGA-PGZ	D-C Power Distribution System (Left-hand radio bay circuit-breaker panel)
XA, XAA-XAZ	A-C Power Supply and Distribution System (Phase "A" circuits)
XC, XCA-XCZ	A-C Power Supply and Distribution System (Phase "C" circuits)
XV, XVA-XVZ	A-C Power Supply and Distribution System (26-volt single-phase circuits)

*Airplanes 139531i through 143593l
†Airplanes 143594m and subsequent

DETERMINING PROPER SYNCHRO CONNECTIONS.

Determining the proper connections of a synchro system when a "crossed wiring" malfunction exists can be accomplished by using figure 6-4A. For example, the synchro transmitter in the adf antenna is set at 0 degrees and its rotation is clockwise. The No. 1 pointer synchro receiver of the radio magnetic course indicator reads 300 degrees and its rotation is also clockwise. In order to determine what connections must be changed to zero the No. 1 pointer receiver and maintain proper rotation, proceed as follows:

a. Place a pencil point at R 300 degrees on figure 6-4A and follow the arrows to S3. This means that reversing the rotor connections to pins "L" and "M" of the radio magnetic course indicator will cause the No. 1 pointer receiver to read 120 degrees.

b. Place a pencil point at S3 120 degrees and note that the shortest distance to 0 degrees is from S3 to S2. This means that connections to pins "N" and "L" of the radio magnetic course indicator must be reversed to make the receiver read 0 degrees. It should be understood that reversing any two stator connections will reverse the direction of rotation; therefore, connections to pins "N" and "P" must now be reversed in order to correct the rotation without changing the zero reading of the receiver.

When using figure 6-4A, always start at the point representing the reading of the synchro receiver (with the synchro transmitter at zero) and follow the arrows to 0 degrees, taking the shortest route. If the starting point on the figure is R 300, R 180, or R 60 degrees, the rotor connections must be reversed in order to arrive at a stator point on the diagram. The appropriate stator connections may then be reversed, if required. To correct rotation only, reverse S3 and S1. If figure 6-4A is used correctly, the following results will be obtained: With transmitter zeroed, the receiver reads as follows: 60 degrees — reverse R1 and R2, S1 and S2; 300 degrees — reverse R1 and R2, S3 and S2; 180 degrees — reverse R1 and R2; 240 degrees — reverse S1 and S2; 120 degrees — reverse S3 and S2.

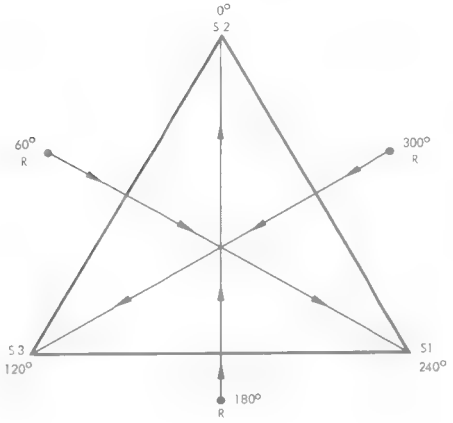


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Figure No. 6-4A. Diagram for Determining Proper Synchro Connections

CONSUMABLE MATERIALS

ITEM NO.	NOMENCLATURE	SPECIFICATION OR STOCK NO.	MANUFACTURER	SUBSTITUTE
9	Anti-seize Compound, White-lead Base, General Purpose (For Threaded Fittings)	JAN-A-669; Stock No. R52C3095		
15	Cement, General Purpose, Synthetic Base	MIL-C-4003		
45	Decalcomanias	MIL-D-8635 (Type II, Class B)		
53	Fluid, Compass, Aircraft	MIL-L-5020; Stock No. R51L395		
90	Methyl-ethyl-ketone	TT-M-261; Stock No. G6810-281-2762		
91	Naphtha, Aliphatic	TT-N-95; Stock No. R52N450		
100	Petrolatum, Technical	VV-P-236; Stock No. WS9150-250-0926		
102	Primer, Zinc Chromate, For Aircraft Use	MIL-P-6889, Type I; Stock No. R52P20660-25		
	Note To obtain the pretinted primer, include the following information: "Pretinted to match interior green, color No. 611." Untinted zinc chromate primer will be applied over wash primer (Specification MIL-C-8514) on surfaces to be painted with lacquer topcoats.			
110	Sealing Compound, General Purpose	NA2-0315	North American Aviation, Inc.	
113	Sealing Compound, Synthetic Glass	MIL-S-7126		
134	Trichlorethylene, Stabilized Degreasing	MIL-T-7003		
140	Cement, High-temperature	No. 32	Sauereisen Cements Co. Pittsburgh 15, Pa.	

INSTRUMENTS

6-1. INSTRUMENTS.

6-2. The instruments are installed in the airplane to give the pilot a quick and certain indication of the attitude, condition or performance of the airplane. The clock, accelerometer and cabin pressure altitude indicator are self-contained instruments and have only to be installed correctly to give an indication. The majority of the instruments, however, are components of indicating systems that reflect conditions at remote parts of the airplane. Keeping all system components as well as the instruments in good operating condition will ensure reliable, useful information which will contribute to the safety of the pilot and the efficient operation of the airplane. Warning lights are installed to indicate unsatisfactory or unsafe conditions. A warning light test switch is provided to check the warning light circuits. Two relays, the warning light dimming relay and the auxiliary warning light dimming relay, are utilized to dim all of the warning lights, with the exception of the fire warning lights and the arresting hook unsafe warning light, whenever the instrument lights rheostat (INSTRUMENTS) is not in the "OFF" position. The approach indexer, installed on airplanes 143594m and subsequent, is also capable of being dimmed (by the auxiliary warning light dimming relay) whenever the instrument lights rheostat (INSTRUMENTS) is not in the "OFF" position. The instruments that require electrical power for operation, or are a part of an electrically remote indicating system, have individual fuses or circuit breakers to isolate the instrument or system in the event of an electrical malfunction.

6-3. INSTRUMENT LOCATION.

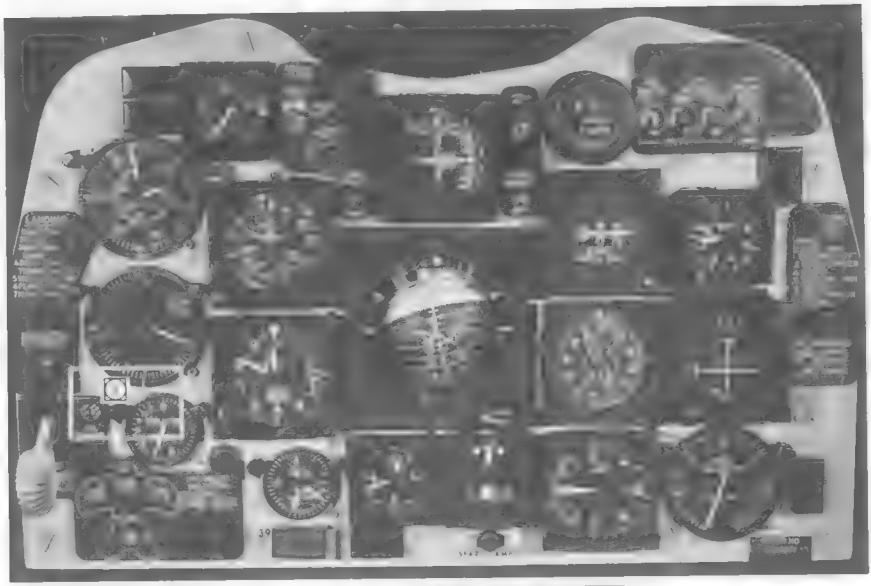
6-4. The majority of the instruments are located on the instrument panel (figures 6-5 and 6-6) and are arranged in groups according to their functions: engine instruments, flight and navigation instruments and miscellaneous instruments. The instruments that are not mounted on the instrument panel are as follows: the stand-by compass which is mounted on the top left-hand side of the instrument panel shroud, the cabin pressure altitude indicator which is located on the right-hand forward vertical console panel adjacent to the instrument panel, the oxygen quantity indicator which is mounted on the left-hand forward vertical console panel adjacent to the instrument panel and the take-off trim position indicator which is located on the left-hand forward console panel. The approach indexer, installed on airplanes 143594m and subsequent, is located on a bracket mounted on the left-hand center windshield defrost line. The bearing converter indicator (if omni-range receiver, AN/ARN-14E, is installed in the airplane), or the azimuth indicator (if radio set, AN/ARN-21, is installed in the airplane) and the statistical accelerometer are mounted in the right-hand radio bay. The crystal current meter is located on the center pedestal and the range meter and on-target light are part of the same assembly which is mounted on the rear of the Mark 8 gun sight. Instrument switches, instrument light switches, instrument system control panels and instrument circuit breakers are located on the left- and right-hand console panels. Instrument circuit breakers and fuses are also located on the right-hand rear vertical console panel in the cockpit and on the left-hand radio bay circuit-breaker panel. The compass and airspeed correction cards are attached to the right-hand side of the canopy in card holders.

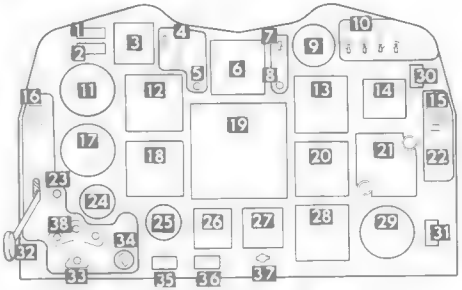
6-5. INSTRUMENT CLASSIFICATION.

6-6. The instruments may be classified or grouped according to their function, principle of operation or source of operating power. The following instruments are installed in the airplane:

INSTRUMENT	PART NUMBER	OPERATING PRINCIPLE	POWER SOURCE
	FLIGHT AND NAVIGATION INSTRUMENTS		
Remote attitude indicator	Lear 978R	Vertical gyro	28 volts dc 115 volts ac
Turn-and-bank indicator	Pioneer Central—1725-IBD-A1-1	Rate gyro	Regulated air pressure
Rate-of-climb indicator	R88-I-0750-110	Diaphragm	Static air pressure
Sensitive altimeter	R88-A-0351-011	Diaphragm	Static air pressure
Airspeed and Mach number indicator	R88-I-0474-100	Diaphragm and aneroid bellows	Pitot-static air pressure
Radio magnetic course indicator	R16-AN ID-250A/ARN	Synchro	26 volts ac
Course indicator	R16-AN ID-249B/ARN	Synchro	26 volts ac
Bearing converter indicator*	R16-AN ID-251/ARN	Synchro	26 volts ac
Azimuth indicator†	R16-AN ID-307/ARN	Synchro	26 volts ac
Range indicator†	R16-AN ID-310/ARN	Synchro	26 volts ac
Stand-by compass	R88-C-0779-525	Magnets	Magnetic
Angle-of-attack indicator	Specialties SLZ 9036	Servo bridge circuit	28 volts dc
Approach indexer	Grimes 20020-1-327	Servo bridge circuit	28 volts dc

*Installed on airplanes equipped with omni-range receiver, AN/ARN-14E
†Installed on airplanes equipped with radio set, AN/ARN-21





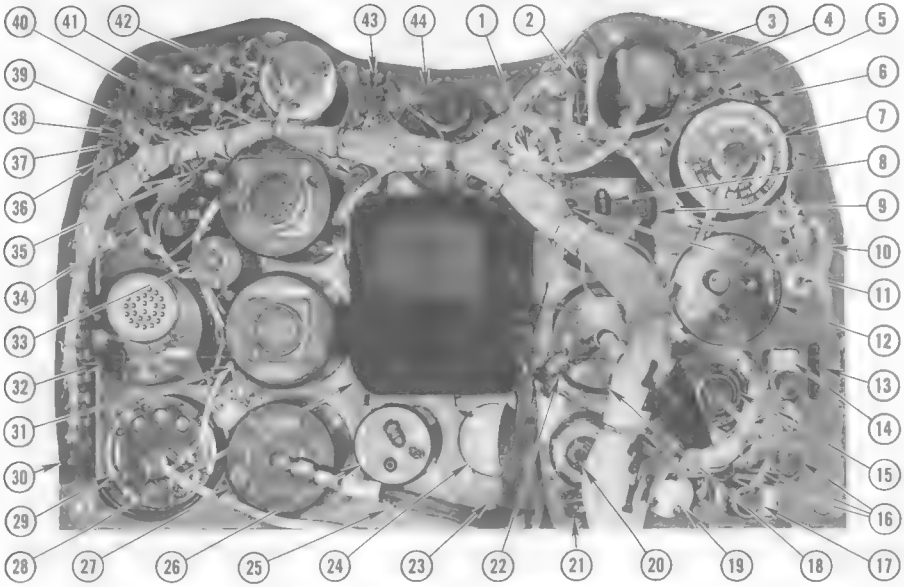
1 FIRE COMPRESSOR WARNING LIGHT
2 FIRE BURNER WARNING LIGHT
3 ANGLE-OF-ATTACK INDICATOR
4 LABS ANGLE SELECTOR SWITCH
5 FORWARD SPEED BRAKES POSITION INDICATOR
6 DIVE-AND-ROLL INDICATOR
7 FIRE DETECTOR TEST SWITCH
8 AFT SPEED BRAKES POSITION INDICATOR
9 LABS TIMER
10 GUN CONTROL SWITCH PANEL
11 TACHOMETER
12 AIRSPEED AND MACH NUMBER INDICATOR
13 RANGE INDICATOR, ID-310/ARN-21
14 ACCELEROMETER
15 TAKE-OFF CHECK LIST
16 LANDING CHECK LIST
17 EXHAUST TEMPERATURE INDICATOR
18 SENSITIVE ALTIMETER
19 REMOTE ATTITUDE INDICATOR
20 RADIO MAGNETIC COURSE INDICATOR
21 COURSE INDICATOR, ID-249B/ARN
22 FUEL GAGE CHECK SWITCH
23 OIL PRESSURE INDICATOR
24 FUEL FLOW INDICATOR
25 HYDRAULIC PRESSURE INDICATOR
26 STANDARD CLOCK
27 TURN-AND-BANK INDICATOR
28 RATE-OF-CLIMB INDICATOR
29 FUEL QUANTITY INDICATOR
30 MANUAL FUEL CONTROL WARNING LIGHT
31 LOW FUEL WARNING LIGHT
32 LANDING GEAR CONTROL HANDLE
33 FLAP POSITION INDICATOR
34 HYDRAULIC PRESSURE SELECTOR SWITCH
35 FLIGHT CONTROL PRESSURE WARNING LIGHT
36 WINDSHIELD ANTI-ICE OVERHEAT WARNING LIGHT
37 SPARE LAMPS
38 LANDING GEAR POSITION INDICATORS

(1) Airplanes 139631 through 143542k

(2) Airplanes 1435431 and subsequent

FJ-4B-2-51-1B

Figure No. 6-5. Pilot's Instrument Panel—Front View



1 FORWARD SPEED BRAKE INDICATOR
2 LABS ANGLE SELECTOR SWITCH
3 ANGLE-OF-ATTACK INDICATOR
4 FIRE COMPRESSOR WARNING LIGHT
5 FIRE BURNER WARNING LIGHT
6 SAFETY CORD FASTENER
7 TACHOMETER
8 PITOT PRESSURE INPUT
9 AIRSPEED AND MACH NUMBER INDICATOR
10 INSTRUMENT PANEL BOND JUMPER
11 STATIC PRESSURE INPUT
12 EXHAUST TEMPERATURE INDICATOR
13 LANDING GEAR CONTROL HANDLE SLOT
14 OIL PRESSURE INDICATOR *
15 FUEL FLOWMETER
16 LANDING GEAR POSITION INDICATORS
17 FLAP POSITION INDICATOR
18 ALTIMETER
19 HYDRAULIC PRESSURE SELECTOR SWITCH
20 HYDRAULIC PRESSURE INDICATOR
21 FLIGHT CONTROL PRESSURE WARNING LIGHT
22 STATIC PRESSURE FLEXIBLE LINES
23 WINDSHIELD ANTI-ICE OVERHEAT WARNING LIGHT
24 STANDARD CLOCK
25 SPARE LAMPS
26 TURN-AND-BANK INDICATOR
27 RATE-OF-CLIMB INDICATOR
28 REMOTE ATTITUDE INDICATOR
29 FUEL QUANTITY INDICATOR
30 TERMINAL STRIP
31 RADIO MAGNETIC COURSE INDICATOR
32 COURSE INDICATOR, ID-249B/ARN
33 INSTRUMENT PANEL VIBRATOR
34 ACCELEROMETER
35 ACCELEROMETER GROUNDING JUMPER
36 INSTRUMENT PANEL STUD FASTENER (TYPICAL SEVEN PLACES)
37 MANUAL FUEL CONTROL WARNING LIGHT
38 SAFETY CORD FASTENER
39 RANGE INDICATOR, ID-310/ARN-21
40 GUN CONTROL SWITCHES
41 AFT SPEED BRAKE INDICATOR
42 LABS TIMER
43 FIRE DETECTOR TEST SWITCH
44 DIVE-AND-ROLL INDICATOR

* On airplanes 1435431 and subsequent, a direct reading oil pressure indicator is provided.

FJ-48-2-51-2A

Figure No. 6-6. Pilot's Instrument Panel—Rear View

INSTRUMENT	PART NUMBER	OPERATING PRINCIPLE	POWER SOURCE
	ENGINE INSTRUMENTS		
Exhaust temperature indicator	Lewis 172 B2	Millivoltmeter	Self-generated
Tachometer	R88-1-2610-000	Synchronous motor	Self-generated
Oil pressure indicator	G. E. 8DJ51-GBL-4*	Electromagnet	28 volts dc
	U. S. Gauge SR-94A or	Synchro	26 volts ac
	Bendix 26800-A65A-1-A3†	Synchro	26 volts ac
Fuel flow indicator	Eclipse-Pioneer		
	25101-B16A-1-B1‡	Synchro	26 volts ac
	R88-1-1206-100§	Synchro	26 volts ac
	MISCELLANEOUS INSTRUMENTS		
Oxygen quantity indicator	Aro 14740	Bridge circuit	28 volts dc 115 volts ac
Fuel quantity indicator	Liquidometer EA856MFC-20*	Bridge circuit	28 volts dc
	Liquidometer EA856MFC-27†		115 volts ac
Landing gear position indicators	G. E. 8DJ51-GBB-4	Electromagnet	28 volts dc
Flap position indicator	G. E. 8DJ51-GBE-4	Electromagnet	28 volts dc
Trim position indicator	G. E. 8DJ51-GBH-4	Electromagnet	28 volts dc
Speed brakes position indicators—			
Forward	G. E. 8DJ51-GBF-4	Electromagnet	28 volts dc
Aft	G. E. 8DJ51-GBF-4	Electromagnet	28 volts dc
Hydraulic pressure indicator	Eclipse-Pioneer 25602-A37C-7-A1	Synchro	26 volts ac
Accelerometer	R88-A-0140-050	Gravitational mass	Gravity
Statistical accelerometer	Maxson M 101A-11-40506070	Gravitational mass	Gravity 28 volts dc
Cabin pressure altitude indicator	R88-1-0625-000	Diaphragm	Cabin air pressure
Standard clock	R88-C-0583-011	Mechanical	Mechanical
Instrument panel vibrator	Globe C-10FA-548	Motor	28 volts dc
On-target light (Refer to paragraph 9-71.)	AN3157-2	Light	28 volts dc
Crystal current meter (Refer to paragraph 9-71.)	International Instruments A-SP157	Meter	APG-30A crystal current
Range meter (Refer to paragraph 9-71.)	International Instruments A-SP161	Meter	
LABS timer [Refer to Supplemental Handbook of Maintenance Instructions (NAVAER 01-60JKE-502A).]	209-510119	Motor	28 volts dc
Dive-and-roll indicator [Refer to Supplemental Handbook of Maintenance Instructions (NAVAER 01-60JKE-502A).]	Minneapolis-Honeywell 433421	Synchro	26 volts ac

*Airplanes 139531i through 143542k
†Airplanes 143543l and subsequent
‡Airplanes 139531i through 139555i
§Airplanes 141444j and subsequent

6-7. REMOVING INSTRUMENTS FROM INSTRUMENT PANEL.

Caution Before removing any instruments, make sure all electrical power to the instruments is turned off.

Note This procedure is typical for the fuel quantity indicator, the fuel flow indicator, the hydraulic pressure indicator, the exhaust temperature indicator, the tachometer indicator and the LABS timer. On airplanes 1435431 and subsequent, the procedure also applied to the oil pressure indicator.

1 Unscrew the tension screw at the lower right of the instrument to loosen the spring-loaded ring clamp assembly.

Note The tension screw for the oil pressure indicator is located at the top right of the instrument.

2 Remove the instrument from the front of the panel by pulling the instrument out of clamp.

3 Disconnect power source from the rear of the instrument case and cover the disconnected plug and receptacle with masking tape.

Note In addition to the power source for the fuel quantity indicator, three color-coded coaxial leads connect to the case. To disconnect these leads, twist counterclockwise.

Caution All exposed electrical plugs and receptacles must be covered to keep out foreign matter that may render the instruments inoperable.

FJ-48-2-51-34

REFRACTOR-COVERED INSTRUMENTS

Note This procedure is typical for the oil pressure indicator, landing gear position indicators, flap position indicator and the two speed brakes position indicators. On airplanes 1435431 and subsequent, this procedure does not apply to the oil pressure indicator.

1 Rotate the instrument panel stud fasteners counterclockwise.

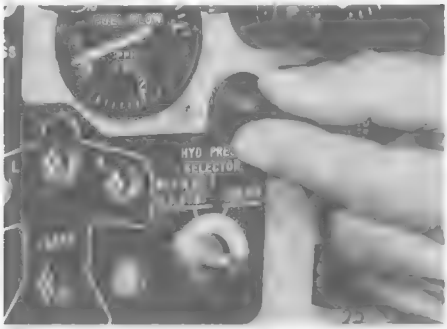
2 Allow instrument panel to come gently aft to position where safety cords are taut.

3 Disconnect the power source from the rear of the instrument case and cover plug and receptacle with masking tape.

4 When removing the oil pressure, landing gear or flap position indicators, remove the hydraulic pressure selector knob.

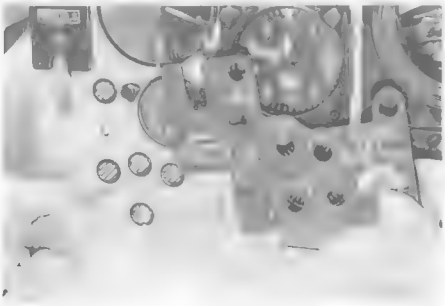
FJ-48-2-51-35

5 Remove the light, or lights, from the refractor covering the instrument to be removed.



6 Remove the refractor from the panel.

7 Remove the two instrument mounting screws from the instrument to be removed.



8 Remove the instrument from the rear of the panel.

REMOTE ATTITUDE INDICATOR

Caution Before removing the remote attitude indicator, make sure all electrical power to the instruments is turned off.

1 Rotate instrument panel stud fasteners counterclockwise and allow the instrument panel to come gently aft to position where safety cords are taut.

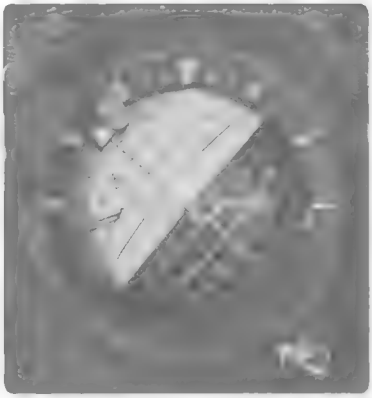
2 Remove clamp from support rod forward of the instrument panel which holds the remote attitude indicator pigtail disconnect.

3 Disconnect the power source.

4 Remove two upper instrument mounting screws and lay aside the lighting fixture.

5 Remove two bottom instrument mounting screws and pull instrument from the panel.

FJ-4B-2-51-44



BEZEL-MOUNTED INSTRUMENTS

Caution Before removing any instruments, make sure all electrical power to the instruments is turned off.

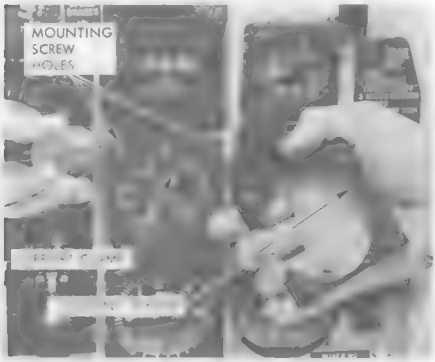
Note This procedure is typical for all of the remaining instruments mounted on the instrument panel.

1 Remove the instrument mounting screw, or screws, that secure the instrument lighting fixture to the case.

FJ-4B-2-51-45

2 Remove the lighting fixture and lay aside. (It is unnecessary to disconnect the wiring from the lights if the same fixture is to be reinstalled.)

3 Remove the two upper instrument mounting screws that secure the support clamp to the instrument case and remove support clamp.



4 Pull the instrument from the panel to expose the power source.

5 Remove the source of power from the rear of the instrument case and cover all disconnected plugs, receptacles or fittings with masking tape.

Note When removing the accelerometer, be careful that the mounting adapters are not pulled loose from the panel.

FJ-48-2-51-46

6-8. INSTALLING INSTRUMENTS IN INSTRUMENT PANEL.

CLAMP MOUNTED INSTRUMENTS

Caution Before installing any instruments, make sure all electrical power to the instruments is turned off.

Note This procedure is typical for the fuel quantity indicator, fuel flow indicator, hydraulic pressure indicator, exhaust temperature indicator, tachometer indicator, LABS timer and, on airplanes 1435431 and subsequent, the oil pressure indicator.

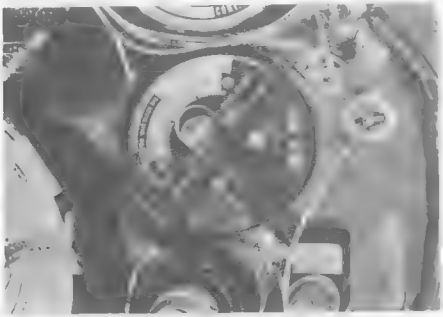
1 Align clamp with panel mounting holes and install and tighten upper left clamp mounting screw. (Clamp mounting screw for the oil pressure indicator mounting clamp is located in the lower left-hand corner.)

2 Install lower right tension screw. (Tension screw for the oil pressure indicator mounting clamp is located in the upper right-hand corner.)

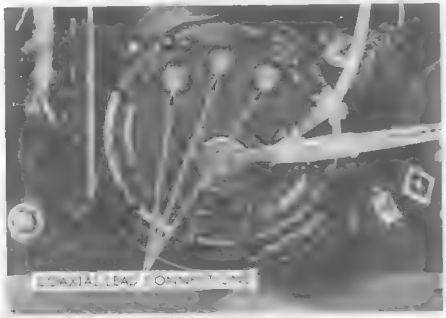
3 Remove masking tape from disconnected plugs, receptacles and/or leads and connect power source to rear of instrument case.

FJ-48-2-51-47

Note On the rear of the exhaust temperature indicator case, the positive and negative connecting studs are of different diameters and are marked "+CR" and "-AL" to ensure proper connection of the two thermocouple leads.

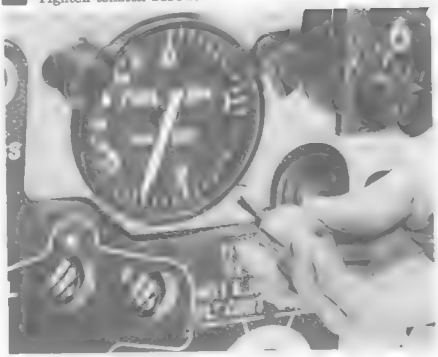


Note In addition to the power source for the fuel quantity indicator, three coaxial leads connect to the rear of the case. To connect these leads, carefully insert lead in receptacle and twist clockwise.



4 Push power source back through the cutout in the panel and insert instrument in panel.

5 Tighten tension screw.



6 Touch up screws with dull black paint.

REFRACTOR COVERED INSTRUMENTS

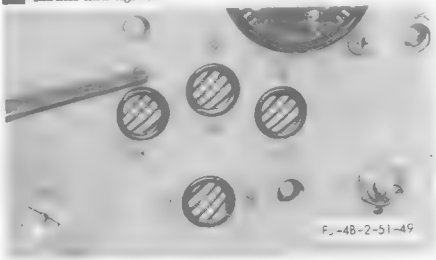
Note This procedure is typical for the landing gear position indicators, flap position indicator, two speed brakes position indicators and, on airplanes 1395311 through 143542k, the oil pressure indicator.

Caution Before installing any instruments, make sure all electrical power to the instruments is turned off.

1 If the instrument panel is not in the servicing position, rotate the seven instrument panel stud fasteners counterclockwise and allow instrument panel to come aft gently to position where safety cords are taut.

2 From the rear of the panel, position the instrument against the cutout provided on the panel.

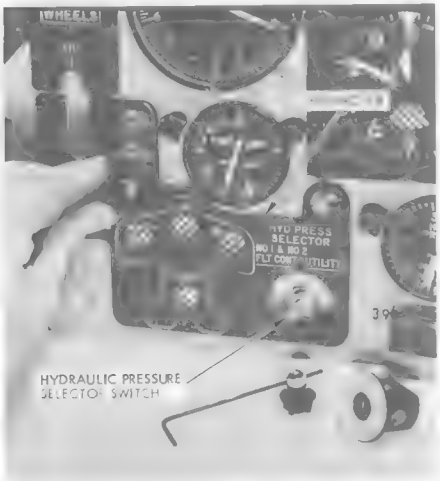
3 Install and tighten the two instrument mounting screws.



4 Remove masking tape from electrical connections and connect power source to rear of instrument case.

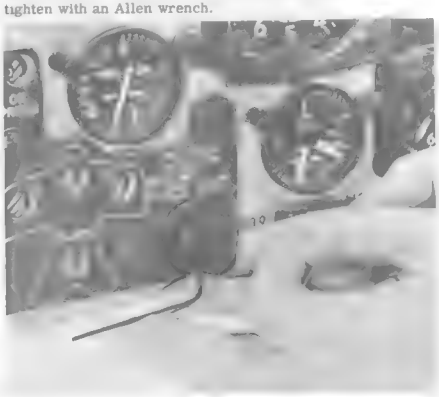
5 Place refractor in position on panel.

6 Install and finger-tighten the light, or lights, securing refractor to panel.



FJ-4B-2-51-50

7 When installing landing gear, flap position or oil pressure indicators, position hydraulic pressure selector knob on hydraulic pressure selector switch and tighten with an Allen wrench.



8 Return instrument panel to its mounted position and secure by turning seven stud fasteners clockwise.

BEZEL MOUNTED INSTRUMENTS

Caution Before installing any instruments, make sure all electrical power to instruments is turned off.

Note This procedure is typical for airspeed and Mach number indicator, sensitive altimeter, rate-of-climb indicator and turn-and-bank indicator.

1 Pull end of flexible hose through cutout in panel. Remove masking tape from fittings.

2 Lubricate threads of male fittings sparingly with anti-seize compound (item 9, materials list).

Caution Do not leave compound on the end of a fitting where it can enter the system and cause malfunctioning of the instrument.

3 Start nut on fitting and turn nut with fingers until it is finger-tight. Do not use a wrench until nut is finger-tight.

4 With a torque wrench, tighten nut to a torque of 70 inch-pounds (minimum torque). Use a backup wrench on flat of fitting so that fitting will not turn when nut is tightened.

FJ-4B-2-51-51

5 Check hose to be sure it is not twisted.

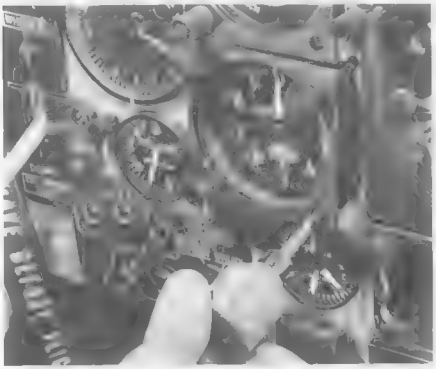
Note Prevent damaging hose or loose connections. Pressure tends to untwist and straighten hose; if hose is twisted it will result in sheared off hose ends or loose fittings.

6 If connection leaks, it may be tightened to a torque of 120 inch-pounds (maximum torque). Use a backup wrench on flat of fitting to prevent fitting from turning.

7 Insert instrument in panel.

8 Position lighting fixture support bracket, or brackets, install and tighten two upper instrument mounting screws.

9 Position lighting fixture and secure with bottom instrument mounting screw, or screws.



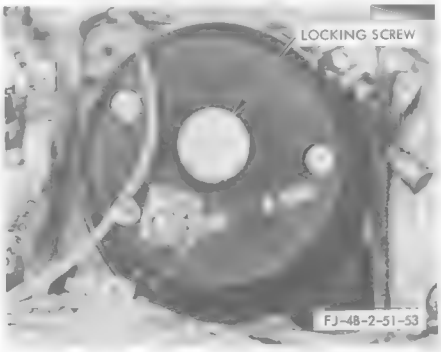
10 Touch up screws with dull black paint.

Note This procedure is typical for angle-of-attack indicator, radio magnetic course indicator, clock, accelerometer, dive-and-roll indicator and course indicator, ID-249B/ARN.

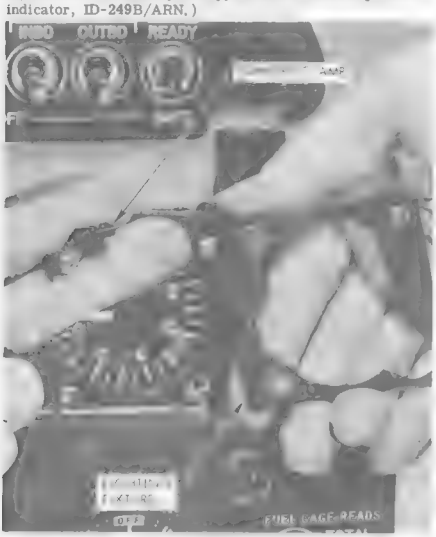
Caution Before installing any instruments, make sure all electrical power to instruments is turned off.

1 Pull power source through cutout on panel. Remove masking tape from disconnected plugs and receptacles and connect power source to rear of instrument case.

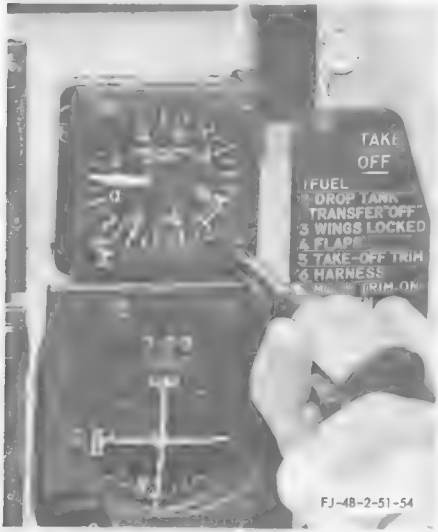
Note No power connections are necessary for the clock and accelerometer. When installing accelerometer, make certain locking screw on rear of case is in free position.



2 Insert instrument in panel. Position lighting fixture support clamp, install and tighten two upper mounting screws. (This step not applicable to installing course indicator, ID-249B/ARN.)



3 Position lighting fixture over instrument dial; install and tighten lower mounting screw, or screws.



FJ-48-2-51-54

Note Course indicator, ID-249B/ARN, has only two instrument mounting screws which secure both the lighting fixture and the instrument to the panel.

4 Touch up screws with dull black paint.

PROCEDURE FOR REMOTE ATTITUDE INDICATOR

1 Insert instrument in panel and install two bottom mounting screws.

2 Position lighting fixture and secure both instrument and lighting fixture with two top mounting screws.

3 If instrument panel is not in servicing position, rotate instrument panel stud fasteners counterclockwise and allow panel to come aft gently to position where safety cords are taut.

4 Connect pigtail to receptacle and safety with AN995F32 lockwire.

5 Clamp connection to support rod forward of instrument panel.

6 Return instrument panel to mounted position and secure with stud fasteners.

FJ-48-2-51-55

7 Touch up top screws with dull black paint.

CEMENTING MOUNTING ADAPTERS

The accelerometer is installed in two mounting adapters which are cemented to the front and rear of the instrument panel. If either adapter becomes loose or detached from the panel during removal or installation, recement adapter to panel as follows:

Note If a rear adapter becomes loose or detached from the panel, it will be necessary to put the instrument panel in servicing position. Rotate the seven instrument panel fasteners counterclockwise, and allow panel to come aft to position where safety cords are taut.

1 Remove all dirt from mounting surfaces by wiping with a cloth dampened with diluent naphtha (item 91, materials list).

2 Brush one even coat of general purpose rubber cement (item 15, materials list) on each of the surfaces to be joined.

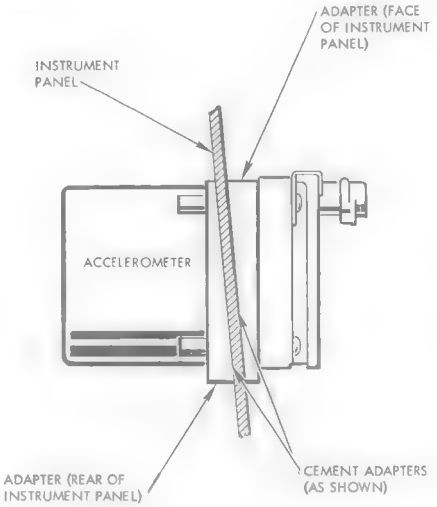
Caution The solvents are flammable and cement must be used with all proper precautions to prevent fires.

Note The cement is relatively fast drying and should not be allowed to stand in open cans. If cement becomes too thick for easy application because of solvent losses, it may be thinned with methyl-ethyl-ketone (item 90, materials list). Do not thin cement beyond its original consistency.

FJ-48-2-51-56

3 Allow cement to dry until it is tacky.

Note To recognize proper stage of tackiness, touch a knuckle to the drying cement. (Do not use fingertip which is likely to be oily.) When cement appears to stick to knuckle but does not adhere to it, it is in proper condition for bonding.



4 Press or roll adapter against panel. This will exclude any bubbles and ensure good adhesion.

Note Do not install instrument for 48 hours after cementing adapter to panel.

5 For cleaning up after cementing, use a cloth dampened with methyl-ethyl-ketone (item 90, materials list). Do not use an excess of this solvent.

FJ-48-2-51-57

6-9. ADJUSTING INSTRUMENTS. The only adjustments that should be made to any instrument installed in the airplane are as follows:

a. Tighten loose screws or connections.

b. Replace mounting screws, washers or nuts.

c. Zero adjustments only on those instruments which have zero setting screws.

d. Special adjustments to an instrument which are specifically described in that respective instrument paragraph.

6-10. REPAIRING INSTRUMENTS. Instruments are delicate precision mechanisms which should never be repaired or tampered with except by a specialist. *No* repair should ever be made on any instrument or on any instrument system unit without specific instructions. If it is suspected that an instrument is defective (or if it has been proved defective through testing), replace the faulty unit with a serviceable one.

6-11. CLEANING INSTRUMENTS. No internal cleaning of an instrument, or of an instrument system unit, should ever be attempted except by an instrument specialist. If internal dirt, foreign objects or discolored parts are visible through the instrument cover glass, replace the instrument. The exterior surfaces of instrument cases and instrument cover glasses may be cleaned when necessary. To clean, obtain three clean, lint-free cloths and proceed as follows:

a. Dip cloth in a solution of mild soap and water and wring until damp.

CAUTION

Make certain the cloth is only damp. Do not wet instrument case or glass.

b. Wipe surfaces thoroughly.

c. Rinse cloth in clear water, wring until damp and again wipe surfaces.

d. Dry immediately with a second, clean, dry cloth.

e. Moisten a third cloth with alcohol and wipe surfaces to remove any soap film. Repeat this procedure on the glass to remove any streaks or dull finish.

6-12. INSTRUMENT LIGHTING.

6-13. All of the instruments located in the cockpit are provided with individual lighting. Ten of the instruments located on the pilot's instrument panel and the oxygen flow regulator (located on the left-hand forward close-out panel) are equipped with individual two-lamp shield-type lighting assemblies. Individual four-lamp shield-type lighting assemblies are used to illuminate the remote attitude indicator and the turn-and-bank indicator. The fuel flow indicator, the fuel quantity indicator, the exhaust gas temperature indicator, the hydraulic pressure indicator and the tachometer are provided with two post-type light assemblies, one near each of the upper corners of the instruments. On airplanes 143543l and subsequent, the oil pressure indicator is also equipped with two post-type light assemblies. One post-type light assembly is installed for the oxygen quantity indicator, the cabin pressure altitude indicator, the range meter and the crystal current meter. Lamps for the shield-type lighting assemblies and the post-type light assemblies may be replaced separately without removing the lighting fixture from the instrument. A spare lamp holder is located on the bottom center of the instrument panel. The stand-by compass and LABS timer each have integral lighting. All of these lights, with the exception of the crystal current meter, are controlled by the instrument lights rheostat (INSTRUMENTS) located on the forward right-hand console panel. The crystal current meter light is controlled by the console lights rheostat (CONSOLE) also located on the forward right-hand console panel. A switch (STANDBY COMPASS & RANGE IND) further controls the lighting of the stand-by compass and the range meter. Twelve light assemblies are installed on refractors on the instrument panel and one assembly is installed on the instrument panel shroud. These lights, which aid in illuminating the position indicators, switches and check lists, are controlled by a rheostat (CONSOLE) located on the right-hand forward console panel. Three

Figure No. 6-7. Instrument Range Markings

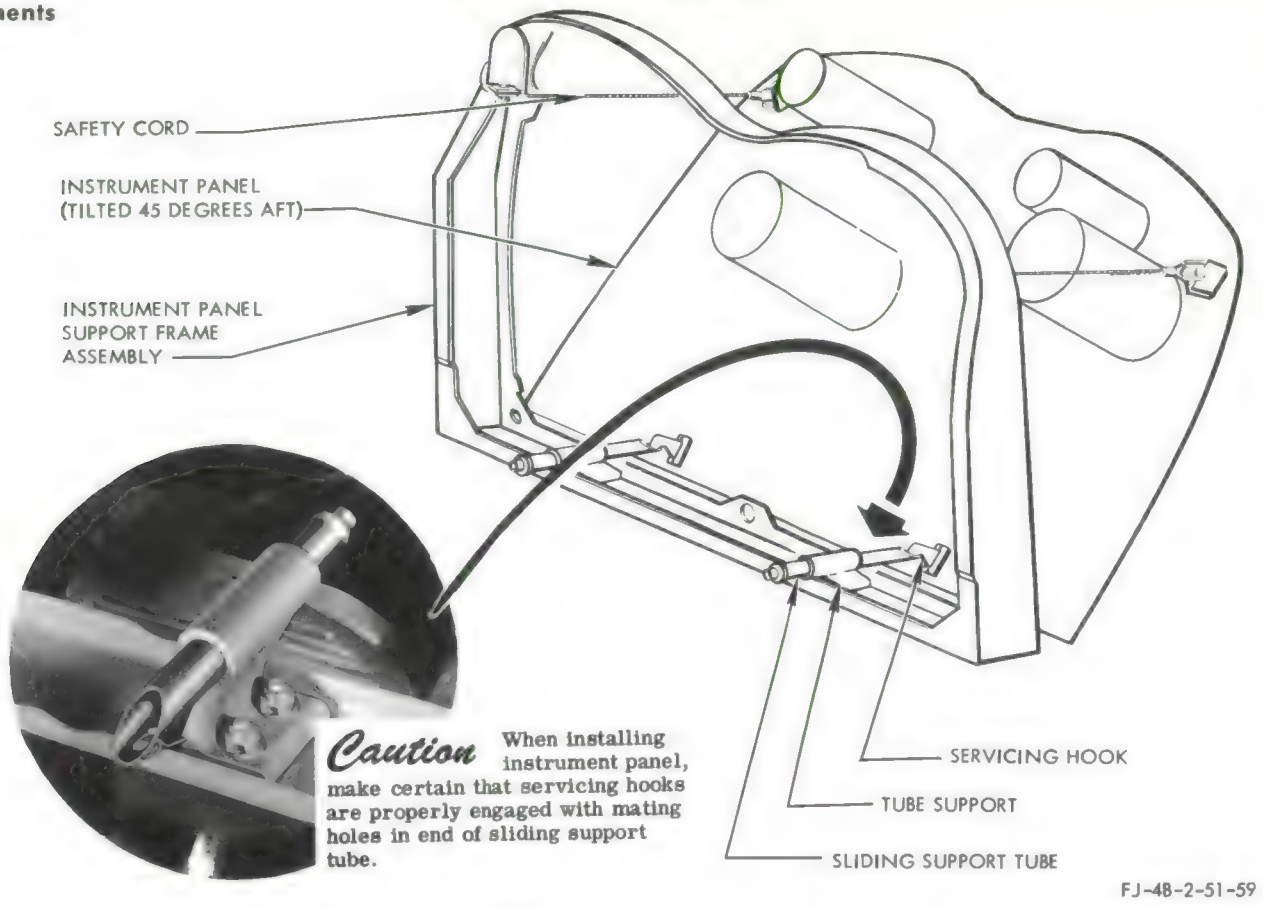


Figure No. 6-8. Instrument Panel Servicing Support Hook and Tube Assembly

instrument floodlights, two on the right side and one on the left side of the instrument panel, direct light at the panel for over-all illumination. The instrument floodlights are controlled by a three-position switch (INST FLOOD) located on the right-hand forward console panel. For further information concerning interior lighting, refer to paragraphs 8-94 through 8-112.

6-14. INSTRUMENT RANGE MARKINGS.

6-15. A long, red radial mark is applied to the bezel of the tachometer at the 100 percent graduation. This danger point indicates the engine's maximum operating speed for War Emergency Power. A short, red radial mark is applied to the bezel of the exhaust temperature indicator at 650°C. This danger point indicates the maximum safe, continuous operating temperature of the exhaust gases in the exhaust tail cone. A red arc from 800°C to 900°C signifies the critical hot start range. (Refer to paragraph 5-16.) The oil pressure indicator on airplanes 1435431 and subsequent is marked with two short red radials. One, at 24 psi, indicates the minimum oil pressure limits and the second, at 40 psi, indicates the maximum oil pressure limits.

6-16. APPLICATION OF INSTRUMENT RANGE MARKINGS. Instrument range markings (figure 6-7) are applied as follows:

a. Clean bezel of instrument where marking is to be applied with diluent naphtha (item 91, materials list).

b. Apply red range marks (item 45, materials list) according to existing instructions. Make certain range marks are positioned exactly as shown in paragraph 6-15 and figure 6-7.

c. (Deleted.)

d. When markings have dried, apply a coat of clear varnish over markings, slightly overlapping all edges.

e. Allow 4 hours for varnish to set before handling.

6-17. INSTRUMENT PANEL.

6-18. The instrument panel is mounted on the instrument panel support frame assembly and is normally secured in an upright position by seven stud fasteners. The frame assembly is adjusted so that the panel, when installed and all connections made, will be inclined forward at an angle, 9 degrees from normal, to the fuselage reference plane. Stabilizer assemblies, or shock absorbers, are attached to the left- and right-hand sides of the instrument panel support frame assembly. Safety cord assemblies are provided in the upper left and right corners of the panel to maintain the panel in a 45-degree aft position for access to the rear of the instruments and for access to equipment forward of the panel. Two instrument panel support hooks, mounted near the lower corners of the panel, are engaged with sliding support tubes for additional support when the panel is in the servicing position. (See figure 6-8.) A shroud across the top of the instrument panel prevents glare on the instrument dials for day flying and reduces glare on the windshield for night flying. Provisions are made for installation of 25 instruments on the instrument panel; all

of the instruments are front-mounted, with the exception of the position indicators which are rear-mounted. On airplanes 139531i through 143542k, the oil pressure indicator is also rear-mounted. An electric vibrator is installed on the rear of the panel to keep the instrument pointers moving freely over the dials. A bonding jumper grounds the instrument panel to the structure of the airplane.

6-19. REMOVING INSTRUMENT PANEL. To remove the instrument panel, proceed as follows:

CAUTION

Make sure all electrical power to the instruments is turned off.

a. Remove the oxygen supply hose from the instrument panel shroud. Move hose away from instrument panel.

b. With an Allen wrench, remove landing gear control knob from landing gear control lever.

c. Rotate the seven stud fasteners on front of the instrument panel counterclockwise. Allow instrument panel to come aft to position where safety cords are taut.

CAUTION

- When the instrument panel has been released, do not allow the panel to fall abruptly to the position where safety cords check the fall. This practice would result in jarring the instruments with possible damage that might render them inoperable.
- When the T-208A control panel or the buddy tanker control panel is installed in the center pedestal, care must be taken to avoid damaging or breaking the warning and indicating lights (located near the top edge of the panels) when allowing the instrument panel to come aft.

d. Remove pitot pressure line from the airspeed and Mach number indicator; remove static pressure input line from airspeed and Mach number indicator. Remove air line from turn-and-bank indicator. Cap all open lines and cap or seal the instrument ports with masking tape.

e. Disconnect electrical connectors from the radio magnetic course indicator, ID-250A/ARN, the course indicator, ID-249B/ARN, the range indicator, ID-310/ARN, and the remote attitude indicator.

f. Disconnect the two thermocouple leads from exhaust temperature indicator.

g. Disconnect bonding jumper from left side of panel.

h. Remove the three coaxial leads from fuel quantity indicator.

i. Disconnect four Cannon plugs located on top of center pedestal. Tie plugs in bags or mask both plugs and receptacles.

CAUTION

All exposed electrical plugs and receptacles, pitot-static lines and fittings and turn-and-bank air lines and fittings must be covered to keep out foreign matter that might cause inaccurate indications or render the instruments inoperable.

j. Remove safety cords from fasteners at upper corners of instrument panel.

CAUTION

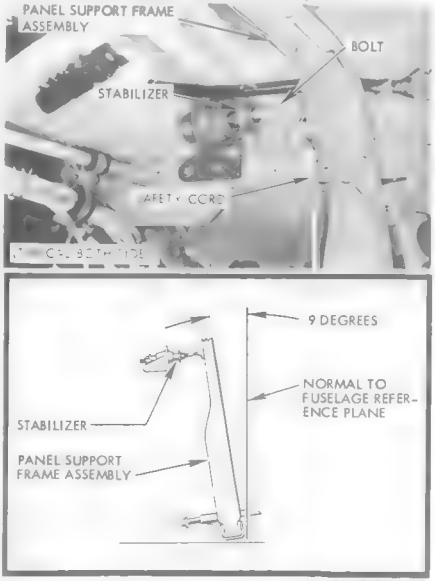
All possible precautions must be taken to prevent instrument panel from falling against the control stick when safety cords are removed from panel.

k. Hold instrument panel securely and lift to disengage the servicing hooks from the sliding support tubes. (See figure 6-8.)

l. Lift instrument panel from the airplane.

6-20. ADJUSTING INSTRUMENT PANEL SUPPORT FRAME ASSEMBLY.

The instrument panel support frame assembly should be adjusted so that the panel is inclined forward 9 degrees from a normal to the fuselage reference plane.



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1 Place an inclinometer fore and aft across the bottom of the instrument panel support frame. Take a reading from aft to forward.

2 Remove the gun bay access door.

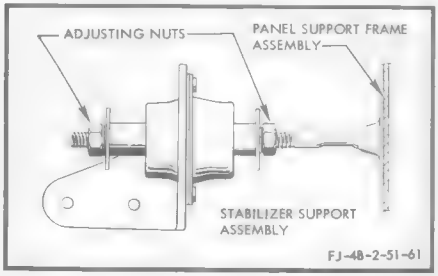
3 With a bubble protractor, take a reading along one of the gun rails. This reading should be to the nearest one-half degree.

Note The gun rails in the gun bay are parallel within one-half degree to the fuselage reference plane.

4 Subtract the second reading from the first.

Note The difference between the first reading and the second reading will be 90 degrees plus or minus the amount of panel inclination.

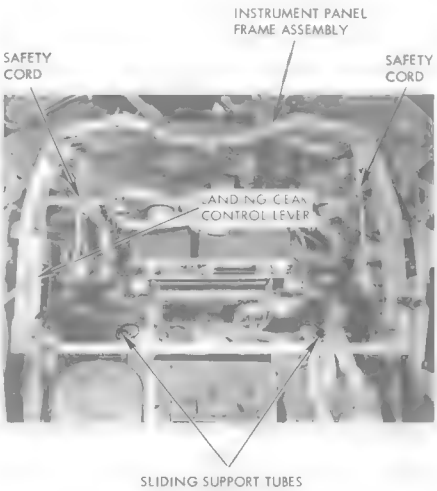
5 If the instrument panel is not inclined forward at an angle of 9 degrees, adjust the angle of the frame by repositioning the nuts on the stabilizer support assembly bolts as necessary.



6-21. INSTALLING INSTRUMENT PANEL.

Caution Before installing the instrument panel, make sure all power is off.

1 Holding the instrument panel securely, position the cutout in the panel over the landing gear control lever and insert both instrument panel servicing support hooks into the mating holes of the sliding support tubes.



2 Connect the two safety cords to the fasteners at the upper left and right corners of the panel.

3 Remove the masking tape from the four receptacles on top of the center pedestal and remove tape or bags from Cannon plugs. Make connections and safety-wire with AN995F32 lockwire.

4 Make electrical connection to remote attitude indicator and clamp the plug and receptacle to support rod forward of the instrument panel.

5 Remove caps from pressure lines and masking tape from instrument fittings. Connect pitot pressure line to pitot port at airspeed and Mach number indicator. Connect static pressure line to airspeed and Mach number indicator and connect air line to turn-and-bank indicator.

6 Connect the three coaxial leads to the fuel quantity indicator. To connect, carefully, but firmly, push the plug into the receptacle and twist clockwise.

7 Connect the two thermocouple leads to the exhaust temperature indicator.

Note The connecting posts on the back of the instrument case are of different diameters and are coded "+CR" and "-AL" to ensure proper connections.

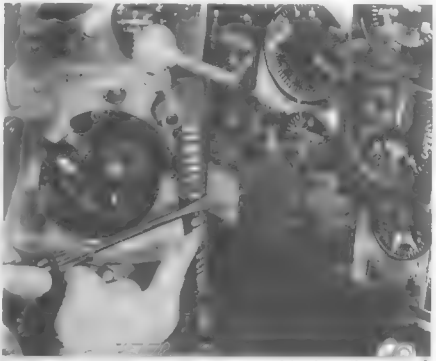
8 Connect the bond jumper to the left of the instrument panel.

9 Remove bags and masking tape from the plugs and receptacles of the radio magnetic course indicator, ID-250A/ARN, the course indicator, ID-249B/ARN and the range indicator, ID-310/ARN and make electrical connections.

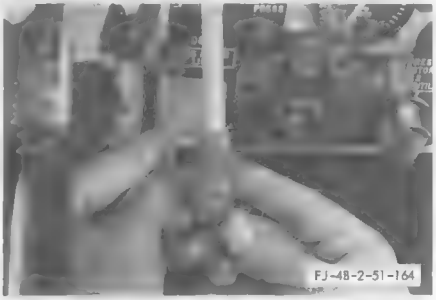
10 Push the instrument panel forward into its mounted position and secure by rotating the seven fasteners on the panel clockwise.

FJ-48-2-51-63

Caution In order to secure the panel as firmly as possible, first tighten the fasteners at the upper right, lower left, upper left and lower right in that order; then, tighten the remaining two upper and the lower center fasteners. Be certain that wire bundles are not pinched by the positioning of the panel. If one or more fasteners do not tighten securely, loosen all fasteners and reseat panel.



11 Position the landing gear control knob on the control lever and tighten the screws with an Allen wrench.



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6-22. CLEANING INSTRUMENT PANEL. To remove dust and dirt from the instrument panel with instruments installed, proceed as follows:

a. Make certain that all instruments are tightly connected to their respective power lines or air lines.

CAUTION

If there is a loose fitting or connection, damage to instruments may occur.

b. Blow loose dust and dirt free of panel by using an air hose. If it is necessary to remove dirt smudges or fingerprints from the instrument panel, refer to paragraph 6-11 and clean as in cleaning instrument cases.

6-23. INSTRUMENT PANEL VIBRATOR.

6-24. A 28-volt d-c miniature motorized instrument panel vibrator (figure 6-9) in installed on the back of the instrument panel to reduce friction within the instrument cases. The vibrator has a speed range between 2400 and 3000 rpm and at 2500 rpm exerts a peak-to-peak amplitude vibration on the instrument panel of 0.003 to 0.005 inch (shaking force of 2.4 pounds). This force is sufficient to keep the pointers moving freely over the dials. Power is from the secondary bus, and a 5-ampere circuit breaker (INST VIBRATOR), located on the left-hand radio bay circuit-breaker panel, protects the circuit. The electrical connections should be checked periodically, but no other adjustments are necessary.

6-25. COCKPIT WARNING LIGHTS.

6-26. The cockpit warning lights (figures 6-10 through 6-21) are provided to indicate unsatisfactory or dangerous conditions and alternate operation of equipment. The cap-covered warning lights, consisting of two standard AN3121-313 lamps, are number coded on the caps and on the instrument panel to ensure the proper lettering of each cap to its respective circuit. All of the warning lights may be tested by depressing the warning light test switch (WARNING LIGHT TEST) located on the right-hand forward console panel. Refer to the following chart for warning light location, lettering, coding and the associated system.

LIGHT	LOCATION	LETTERING	NUMBER CODING	ASSOCIATED SYSTEM
Generator Failure	Right-hand Console	GEN OUT	17	D-C Generator
Inverter Failure	Right-hand Console	INST PWR OFF	3	A-C Instrument Power
Arresting Hook	Arresting Hook Control Handle	None	None	Arresting Hook
Landing Gear	Landing Gear Control Handle	None	None	Landing Gear
Low Fuel	Pilot's Instrument Panel	LOW FUEL	15	Fuel Quantity Indicating System
Manual Fuel Control	Pilot's Instrument Panel	MANUAL FUEL CONTROL	41	Manual Fuel Control
Windshield Anti-ice Overheat	Pilot's Instrument Panel	WINDSHIELD ANTI-ICE OVERHEAT	13	Windshield Anti-Ice
No. 1 and No. 2 Flight Control Pressure	Pilot's Instrument Panel	FLIGHT CONTROL PRESSURE	39	Flight Control, Electrical and Hydraulic
Compressor Fire	Pilot's Instrument Panel	FIRE COMPRESSOR	9	Engine Fire Detector
Burner Fire	Pilot's Instrument Panel	FIRE BURNER	11	Engine Fire Detector
Low Oxygen	Left-hand Forward Close-out Panel	None	None	Oxygen Quantity Indicating System

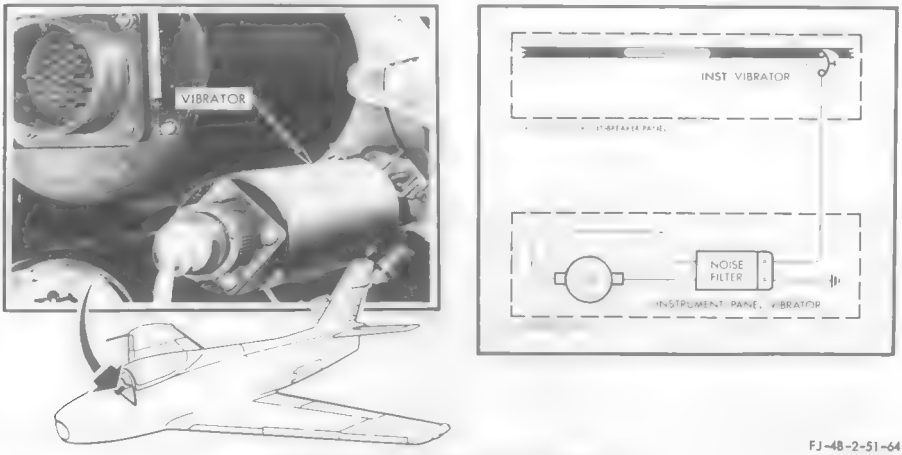


Figure No. 6-9. Instrument Panel Vibrator

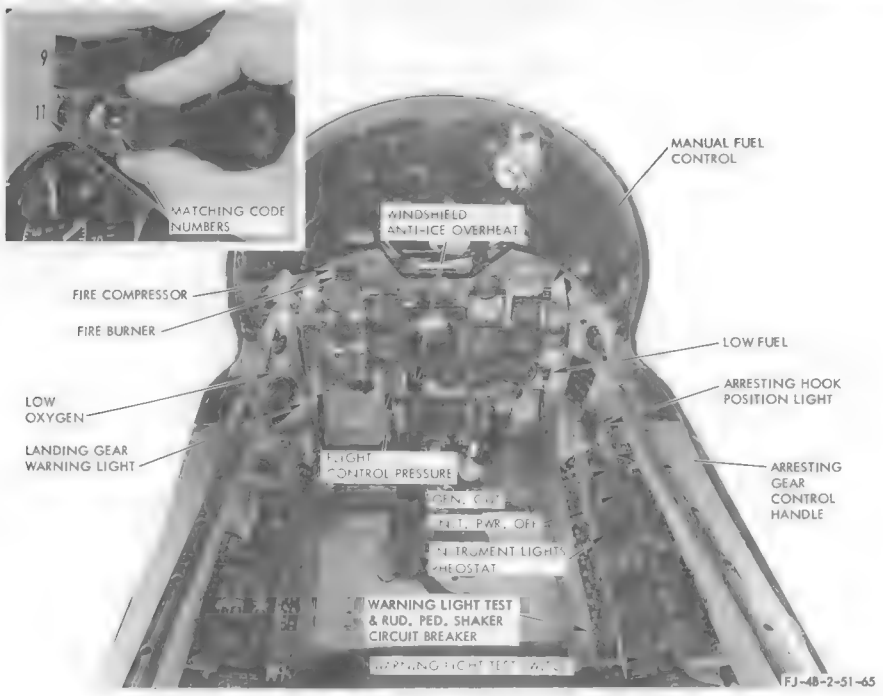


Figure No. 6-10. Cockpit Warning and Indicating Lights

6-27. WARNING LIGHT DIMMING RELAYS.

6-28. The warning light dimming relays (figure 6-11) automatically dim all of the warning lights with the exception of the fire compressor, fire burner and arresting hook unsafe warning lights, whenever the instrument lights rheostat (INSTRUMENTS) is not in the "OFF" position. The approach indexer, installed on airplanes 143594m and subsequent, is also capable of being dimmed whenever the instrument lights rheostat (INSTRUMENTS) is not in the "OFF" position. This reduces glare and prevents the pilot from becoming blinded should one of the warning lights come on during a night flight. The warning light dimming relay receives power from the primary d-c bus and, when energized, directs current flow through a 63-ohm resistor (one resistor for each warning light circuit is contained within the relay), thus dimming the warning lights. The auxiliary warning light dimming relay is installed to dim the low oxygen warning light and, on airplanes 143594m and subsequent, the approach indexer. This relay contains no internal resistors but is mounted with a 500-ohm resistor to dim the low oxygen warning light. On airplanes 143594m and subsequent, three additional 500-ohm resistors (one for each bulb of the approach indexer) are installed to dim the approach indexer. The function of the auxiliary warning light dimming relay is the same as that of the warning light dimming relay. For removing and installing the warning light dimming relays, refer to paragraphs 8-108, 8-109, 8-111 and 8-112.

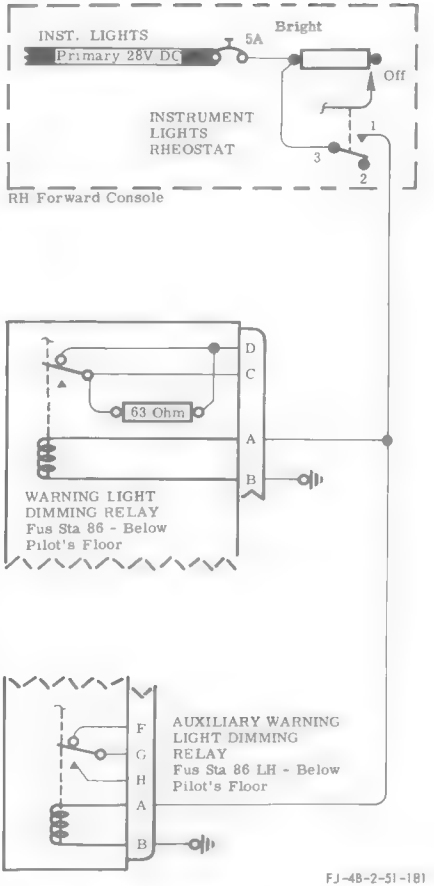


Figure No. 6-11. Warning Light Dimming Relays Schematic

6-29. GENERATOR-OUT WARNING LIGHT.

6-30. The generator-out warning light (GEN OUT) is located on the right-hand forward console panel. (See figure 6-12.) The generator-out warning light circuit is completed and the warning light illuminated whenever the bus control relay is de-energized. Whenever the generator output is connected to the primary d-c bus, the bus control relay becomes energized, the warning light circuit is broken and the warning light goes out.

6-31. INSTRUMENT POWER OFF WARNING LIGHT.

6-32. The instrument power off warning light (INST PWR OFF), also located on the forward right-hand console panel, illuminates whenever an a-c power failure occurs. (See figure 6-13.) An a-c power failure causes the inverter failure indicating relay to become de-energized, completing a circuit to the instrument power off warning light. If sufficient power is available with the a-c power switch in the opposite position, the inverter failure indicating relay will again be energized and the instrument power off warning light will go out.

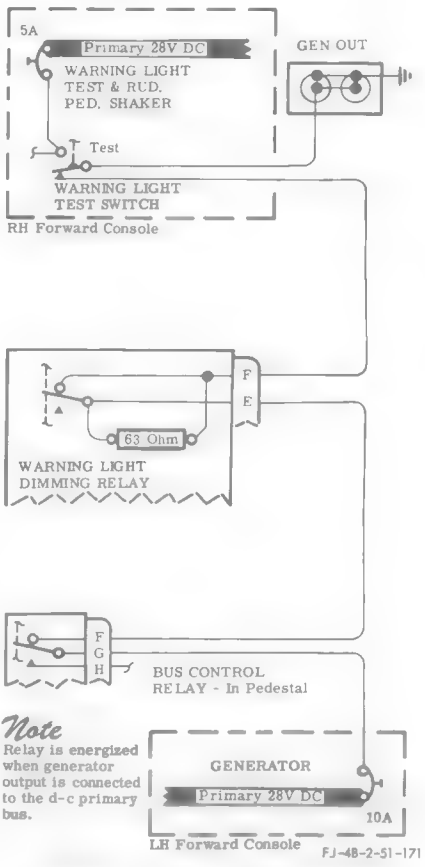


Figure No. 6-12. Generator-out Warning Light Schematic

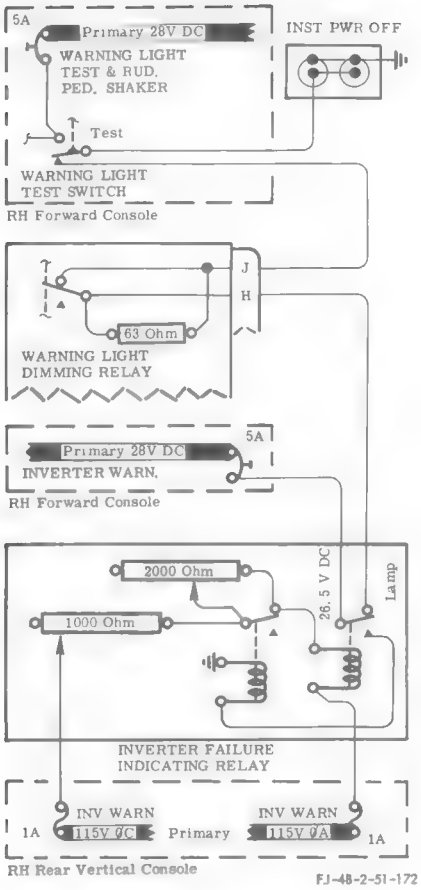


Figure No. 6-13. Instrument Power Off Warning Light Schematic

6-33. LANDING GEAR WARNING LIGHT.

6-34. The landing gear warning light (figure 6-14) is a single red light located in the landing gear control handle. The light will illuminate whenever an unsafe landing gear condition exists. When the landing gear control handle is raised, the light will come on and remain on until the gear is up and locked and the wheel and strut doors are closed and locked. When the landing gear handle is lowered, the light will come on and remain on until the landing gear is down and locked. On the down cycle, the light will go out regardless of the position of the wheel and strut doors.

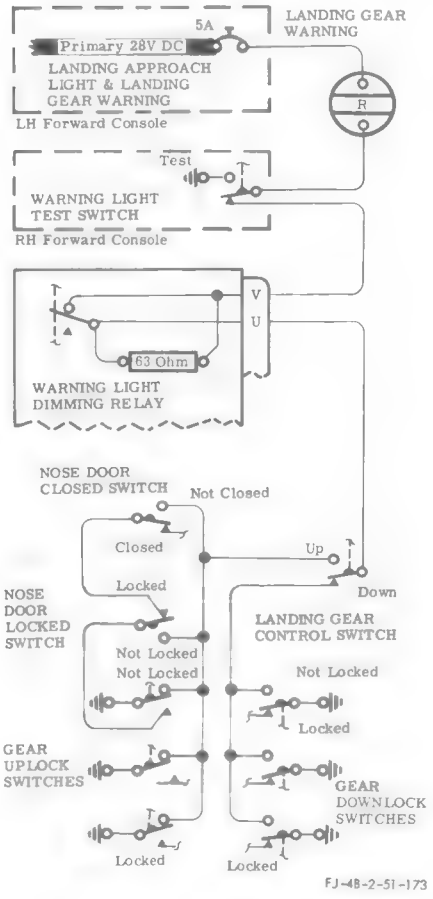


Figure No. 6-14. Landing Gear Warning Light Schematic

6-35. LOW OXYGEN WARNING LIGHT.

6-36. The low oxygen warning light (figure 6-15) is located on the left-hand forward close-out panel. A switch, contained within the oxygen quantity indicator, closes when an indication of 0.5 (±0.1) liter appears on the face of the indicator. The closed contacts of this switch complete the circuit to the low oxygen warning light and illuminate the light.

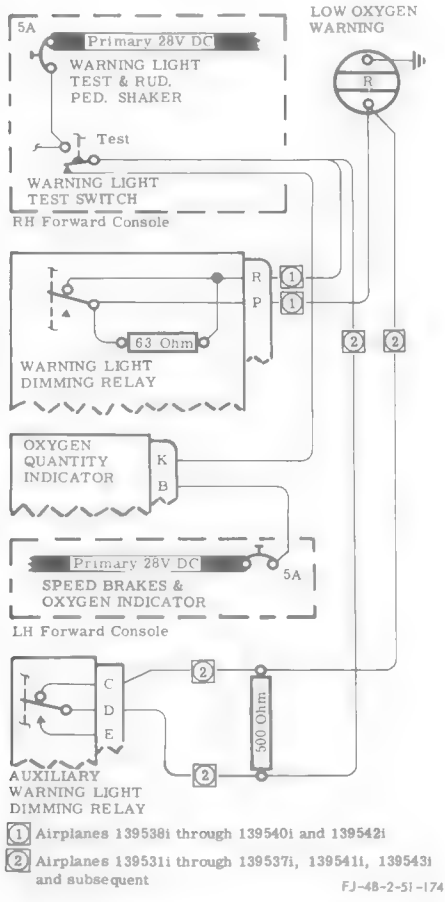


Figure No. 6-15. Low Oxygen Warning Light Schematic

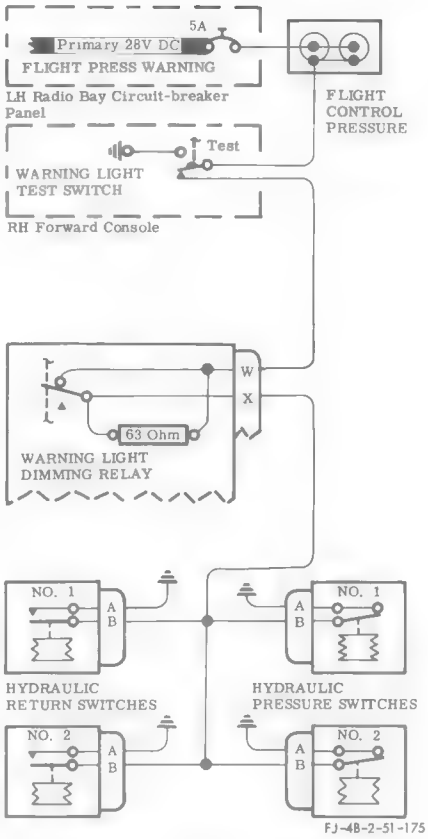


Figure No. 6-16. Flight Control Pressure Warning Light Schematic

6-37. FLIGHT CONTROL PRESSURE WARNING LIGHT.

6-38. The flight control pressure warning light (FLIGHT CONTROL PRESSURE) works in conjunction with four hydraulic pressure switches, two of which are installed in the flight control pressure lines and two of which are installed in the flight control return lines. (See figure 6-16.) The warning light illuminates whenever hydraulic pressure in the flight control pressure lines drops below 650 (±50) psi, or whenever hydraulic pressure in the flight control return lines exceeds 1750 (±100) psi.

6-39. MANUAL FUEL CONTROL WARNING LIGHT.

6-40. The manual fuel control warning light (MANUAL FUEL CONTROL) is illuminated whenever the MANUAL FUEL CONT switch, located on the left-hand forward console panel, is in the "MANUAL" position. Placing the switch in the "PRIMARY" position will extinguish the light. (See figure 6-17.)

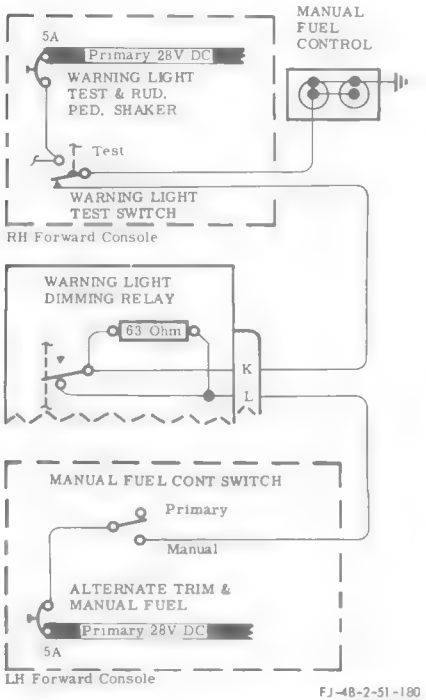


Figure No. 6-17. Manual Fuel Control Warning Light Schematic

6-41. LOW FUEL WARNING LIGHT.

6-42. A low fuel level limit switch in the forward fuselage fuel cell opens when the fuel level drops to 950 (±50) pounds. When this switch is open, the low fuel sensing relay, located in the right-hand radio bay electrical junction box, is de-energized, thus providing a ground to the low fuel warning light (LOW WARNING) and causing the light to illuminate. (See figure 6-18.)

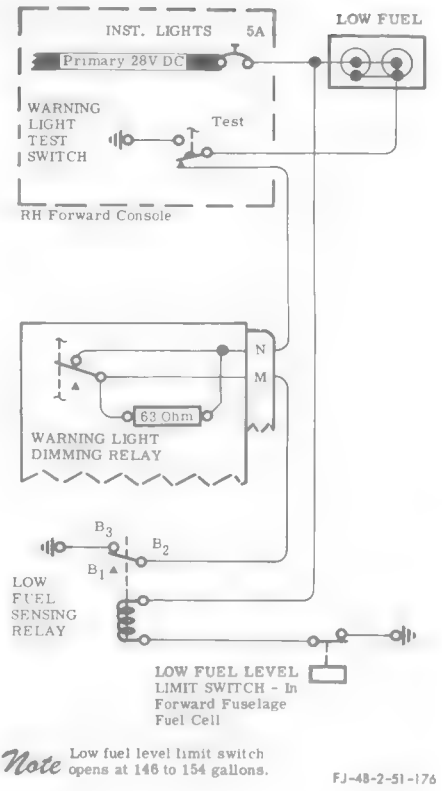


Figure No. 6-18. Low Fuel Warning Light Schematic

6-43. WINDSHIELD ANTI-ICE OVERHEAT WARNING LIGHT.

6-44. If the windshield anti-ice supply air becomes too hot, the windshield may be cracked. To warn the pilot of an overtemperature condition, the windshield anti-ice overheat warning light (WINDSHIELD ANTI-ICE OVERHEAT) is provided. (See figure 6-19.) The windshield overheat detector is a thermoswitch which closes when the supply air temperature reaches 300 (±8)°F, completing the circuit to the warning light.

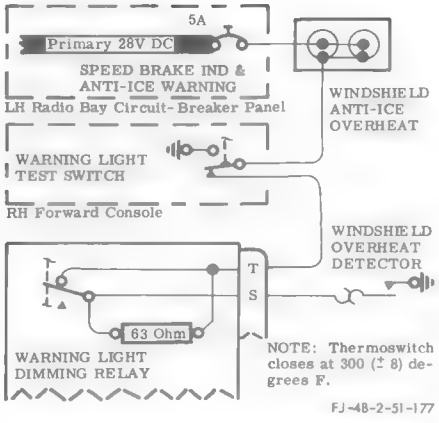


Figure No. 6-19. Windshield Anti-ice Overheat Warning Light Schematic

6-45. FIRE COMPRESSOR AND FIRE BURNER WARNING LIGHTS.

6-46. The fire compressor and fire burner warning lights (FIRE COMPRESSOR and FIRE BURNER) are a part of two separate fire detector systems. (See figure 6-20.) The compressor fire detector system will indicate a fire or overtemperature condition in the forward fuselage near the engine compressor section, and the burner fire detector system will indicate a fire or overtemperature condition in the aft fuselage near the engine burner section. These systems are made up of lengths of sensing element which will illuminate the warning lights if any one foot of sensing element is exposed to temperatures in excess of 685 (±35)°F.

6-47. ARRESTING HOOK UNSAFE WARNING LIGHT.

6-48. The arresting hook unsafe warning light (figure 6-21) is located in the arresting hook control handle. The arresting hook must be either fully retracted or fully extended or the warning light will be illuminated.

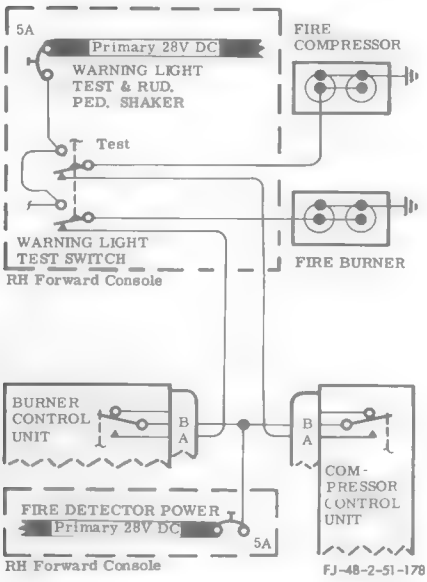


Figure No. 6-20. Fire Compressor and Fire Burner Warning Lights Schematic

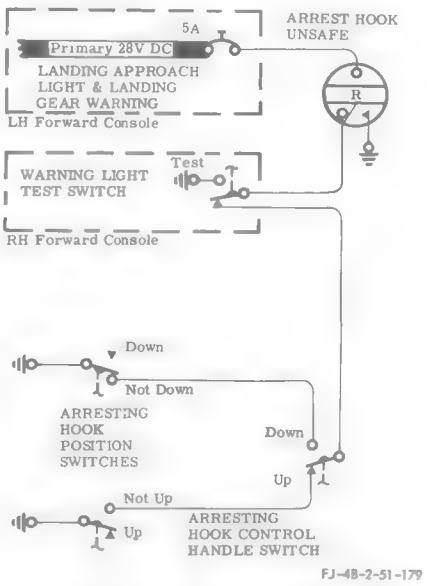


Figure No. 6-21. Arresting Hook Unsafe Warning Light Schematic

6-48A. APPROACH INDEXER DIMMING CIRCUIT.

6-48B. The approach indexer, installed on airplanes 143594m and subsequent, is wired through the auxiliary warning light dimming relay and three 500-ohm resistors to provide a dimming circuit for each of the three bulbs it contains. The approach indexer dimming circuit is controlled by the instrument lights rheostat (INSTRUMENTS). Whenever the INSTRUMENTS rheostat is in the "OFF" position, power, taking the path of least resistance, flows through contacts of the de-energized auxiliary warning light dimming relay and on to the bulbs of the approach indexer, providing a bright indication. When the INSTRUMENTS rheostat is turned out of the "OFF" position, the auxiliary warning light dimming relay becomes energized, allowing no power to flow through it. Current is then forced to flow through the 500-ohm resistors, thus dimming the bulbs of the approach indexer. See figure 6-21A for a schematic of the approach indexer dimming circuit.

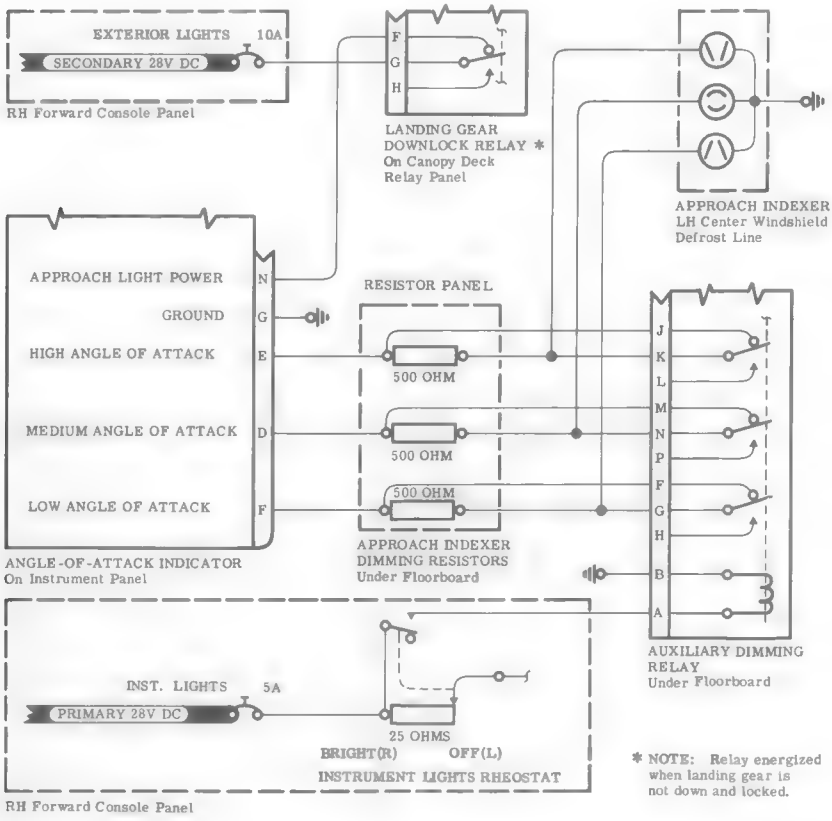


Figure No. 6-21A. Approach Indexer Dimming Schematic—Airplanes 143594m and Subsequent

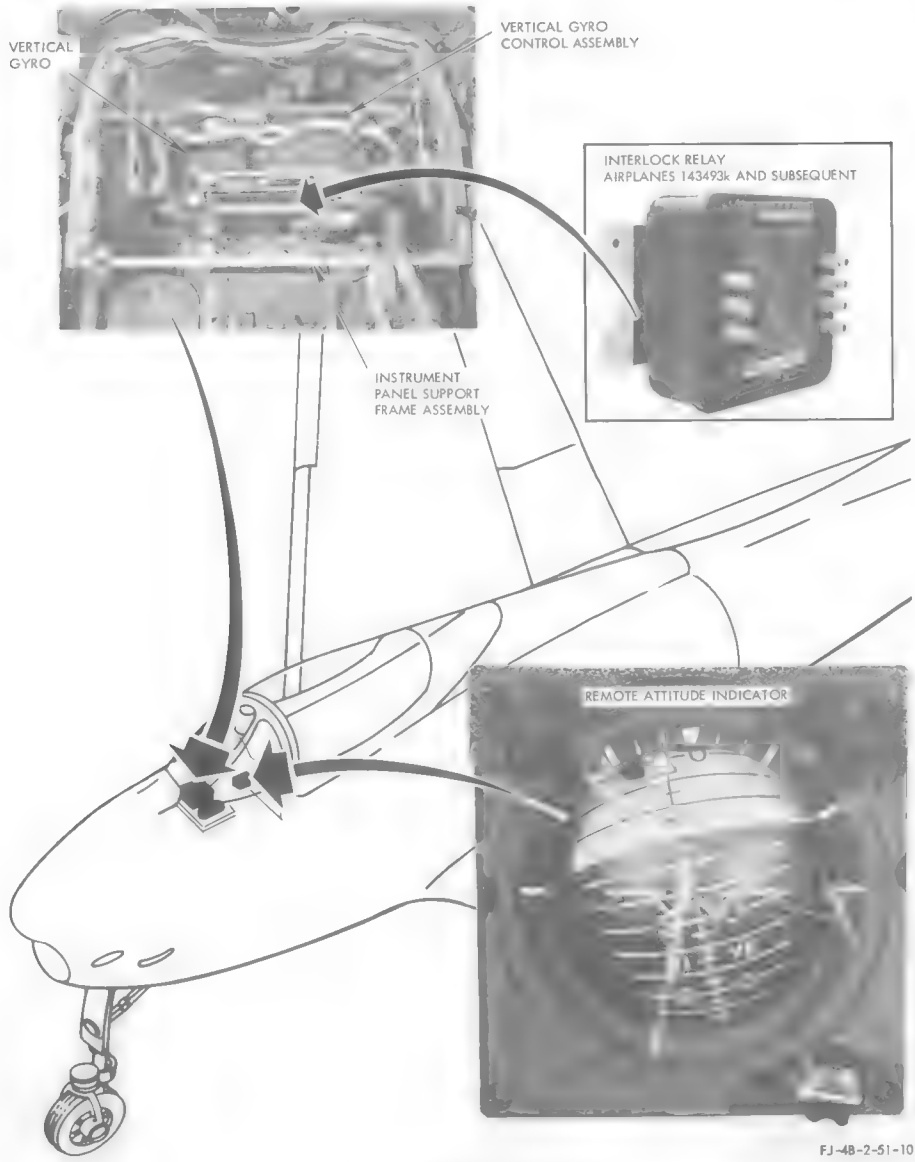


Figure No. 6-22. Vertical Gyro System

VERTICAL GYRO SYSTEM

6-49. VERTICAL GYRO SYSTEM.

6-50. The vertical gyro system (figures 6-22 and 6-23) consists of a vertical gyro and a vertical gyro control assembly, which are located on a support bracket mounted on the forward bulkhead of the cockpit area, and a remote attitude indicator located on the pilot's instrument panel. An a-c—d-c power interlock relay is installed on airplanes 143493k and subsequent and is located under the support bracket of the vertical gyro and vertical gyro control assembly. The vertical gyro system provides attitude reference indications which are accurate within one-half of one degree in straight and level flight and within 2 degrees or less in turns and maneuvers. The gyro rotor in the vertical gyro unit is a 115-volt, 400-cycle, 3-phase induction motor which rotates at a speed of approximately 22,000 rpm. The spin axis of the gyro rotor establishes a vertical reference line from which roll and pitch deviations are measured. A small position transmitter is placed in the roll axis and another in the pitch axis of the vertical gyro. Angular displacements, with respect to the vertical reference, are measured electrically by means of the position transmitter synchros and the receiver synchros located in the attitude indicator. The voltage produced at the receiver synchro is fed through an amplifier to a servomotor located in the indicator. The indicating system receives 115-volt, 400-cycle, alternating current from the primary phase "A" and phase "C" busses and 28-volt, direct current from the primary d-c bus. The a-c circuits are protected by one-ampere fuses (GYRO HORIZ), located on the right-hand rear vertical console, and the d-c circuit is protected by a 5-ampere circuit breaker (GYRO HORIZON) located on the right-hand forward console panel.

6-51. FUNCTION OF VERTICAL GYRO SYSTEM.

6-52. During straight and level flight, the transmitter synchro and the receiver synchro, located in the pitch axis indicating system (figure 6-24), are in an electrically balanced or null condition and the sphere in the attitude indicator is centered. When the airplane noses down or up, the receiver synchro and the indicator rotate with the airplane around the pitch axis. The secondary coil (stator) of the transmitter synchro, which is attached to the gimbal of the gyro, also rotates with the airplane around the gyro-stabilized primary coil (rotor). This action produces a voltage at the receiver synchro secondary coil which is fed into the amplifier channel and to the pitch servomotor assembly in the indicator. The torque of the motor drives the sphere to a new position. The pitch servomotor also mechanically repositions the secondary coil of the receiver synchro and the sphere in the indicator until the synchro circuit is at a null. The sphere has thus moved up or down with respect to the fixed miniature airplane to indicate a dive or climb attitude. When the airplane returns to straight and level flight, a similar action takes place to return the sphere to its indicated level-flight position. The indicating system is damped to eliminate overshoot or hunting at the indicator by the addition of a velocity generator follow-up action. The rotor of the velocity generator is mounted on the pitch axis servomotor shaft, and the output of the generator is connected, through a transformer, in series with the output of the receiver synchro secondary coil which is mounted on the same shaft. When the servomotor moves as a result of the amplified output signal of the receiver synchro secondary coil, the generator also starts to move. Rotation of the generator produces an output signal with the same frequency as the receiver synchro but 180 degrees out of phase. The roll axis indicating system is similar to the pitch axis indicating system, except the transmitter synchro secondary coil is attached to the frame of the gyro and the primary coil is attached to the gimbal of the vertical gyro.

6-53. TROUBLE SHOOTING VERTICAL GYRO SYSTEM—AIRPLANES 139531i THROUGH 139555i AND 141444j THROUGH 141489j.

TEST EQUIPMENT: D-C voltmeter.
A-C voltmeter.
Phase sequence tester. (See figure 6-36.)

SYSTEM CONDITIONS: 28-Volt d-c power applied to airplane.
GYRO HORIZON and NO. 1 INV PWR circuit breakers engaged.
INST. AC POWER switch in "NO. 1 INV." position.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
REMOTE ATTITUDE INDICATOR DOES NOT OPERATE AND POWER FAILURE FLAG DOES NOT LIFT FROM VIEW.			
D-C power fault to vertical gyro control assembly.	Check between test point F and ground.	28 volts dc.	No action.
		Zero volts.	Replace defective d-c power wire.

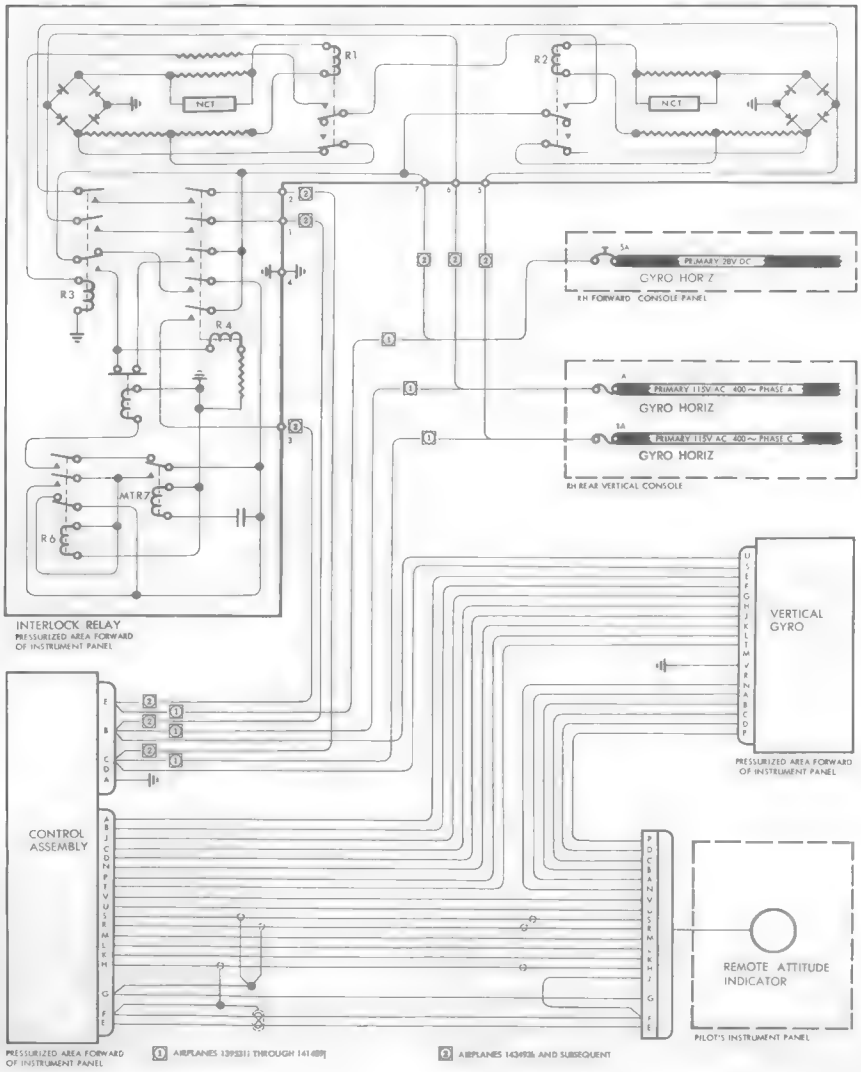


Figure No. 6-23. Vertical Gyro System Schematic

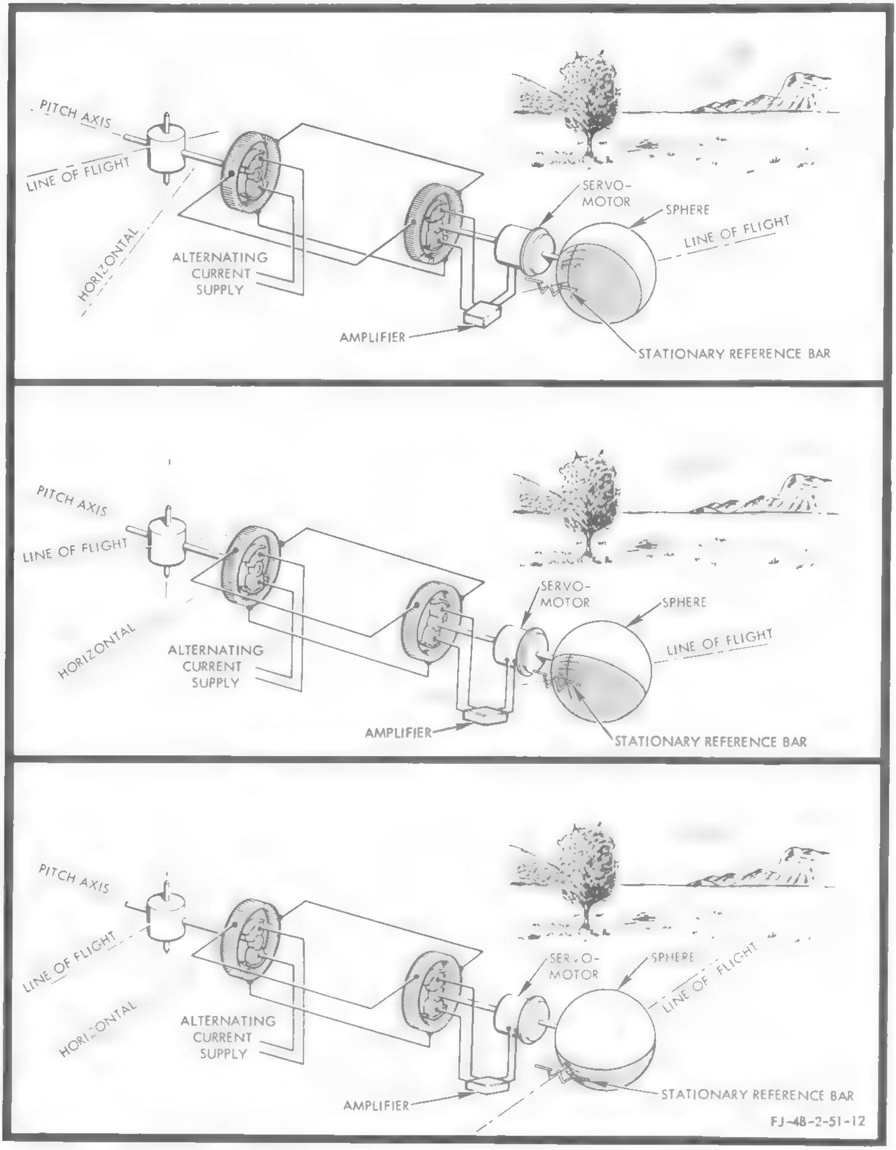


Figure No. 6-24. Pitch Axis Indicating System

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
REMOTE ATTITUDE INDICATOR DOES NOT OPERATE AND POWER FAILURE FLAG DOES NOT LIFT FROM VIEW. (Cont)			
A-C power fault to vertical gyro control assembly.	Check between test points FA and FB and ground.	115 volts ac.	No action.
		Zero volts.	Replace defective a-c power wire(s).
Vertical gyro control assembly a-c input phases "A" and "C" reversed.	Connect phase sequence tester to test points FA, FB and FC.	Tester lamp illuminated.	No action.
		Tester lamp not illuminated.	Reverse wires as necessary.
A-C AND/OR D-C POWER FAILURE.			
Circuit-breaker failure.	Check between test point PBB and ground.	28 volts dc.	Replace GYRO HORIZON circuit breaker.
		Zero volts.	Refer to paragraph 8-61, Trouble Shooting D-C Power Distribution System.
Fuse failure.	Check between test points XAC and XCA and ground.	115 volts ac.	Replace GYRO HORIZ fuse(s).
		Zero volts.	Refer to paragraph 8-78, Trouble Shooting A-C Power Supply and Distribution System.

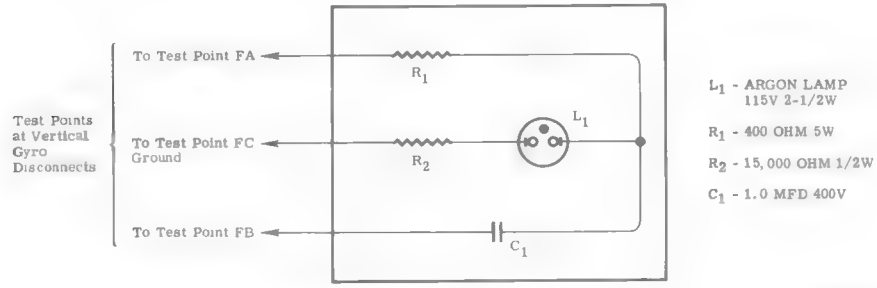
6-53A. TROUBLE SHOOTING VERTICAL GYRO SYSTEM—AIRPLANES 143493k AND SUBSEQUENT.

TEST EQUIPMENT: D-C voltmeter.
A-C voltmeter.
Phase sequence tester. (See figure 6-36.)

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
GYRO HORIZON and NO. 1 INV PWR circuit breakers engaged.
INST. AC POWER switch in "NO. 1 INV." position.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
REMOTE ATTITUDE INDICATOR DOES NOT OPERATE AND POWER FAILURE FLAG DOES NOT LIFT FROM VIEW.			
D-C power fault to vertical gyro control assembly.	Check between test point F and ground.	28 volts dc.	No action.
		Zero volts.	Continue trouble shooting.
A-C power fault to vertical gyro control assembly.	Check between test points FA and FB and ground.	115 volts ac.	No action.
		Zero volts.	Continue trouble shooting.
Defective power interlock relay or associated wiring.	Check between test point FD and ground.	28 volts dc.	Replace defective power interlock relay (paragraph 6-65) or defective d-c power wire to vertical gyro control assembly.
		Zero volts.	Replace defective d-c power wire.
	Check between test points FE and FF and ground.	115 volts ac.	Replace defective power interlock relay (paragraph 6-65) or defective a-c power wire to vertical gyro control assembly.
		Zero volts.	Replace defective a-c power wire.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
REMOTE ATTITUDE INDICATOR DOES NOT OPERATE AND POWER FAILURE FLAG DOES NOT LIFT FROM VIEW. (Cont)			
Vertical gyro control assembly a-c input phases "A" and "C" reversed.	Connect phase sequence tester to test points FA, FB and FC.	Tester lamp illuminated.	No action.
		Tester lamp not illuminated.	Reverse wiring, as necessary.
A-C AND/OR D-C POWER FAILURE.			
Circuit-breaker failure.	Check between test point PBB and ground.	28 volts dc.	Replace GYRO HORIZON circuit breaker.
		Zero volts.	Refer to paragraph 8-61, Trouble Shooting D-C Power Distribution System.
Fuse failure.	Check between test points XAC and XCA and ground.	115 volts ac.	Replace GYRO HORIZ fuse(s).
		Zero volts.	Refer to paragraph 8-78, Trouble Shooting A-C Power Supply and Distribution System.



FJ-48-2-51-204

Figure No. 6-24A. Vertical Gyro Phase Sequence Tester

6-54. OPERATIONAL CHECK OF VERTICAL GYRO SYSTEM.

6-55. (Deleted.)

a. (Deleted.)

b. Place D.C. POWER switch, located on the right-hand forward console, in "OFF" position.

c. Place INST. AC POWER switch, located on the right-hand forward console, in "NO. 1 INV." position.

d. Check that the NO. 1 INV PWR circuit breaker, located on the top deck circuit-breaker panel, is depressed.

e. Check that the GYRO HORIZON circuit breaker, located on the right-hand forward console panel, is depressed.

f. Connect a source of external power to the airplane.

g. Observe the action of the indicator during the warm-up period. Pitch axis erection starts immediately; roll axis erection should start 15 seconds later.

h. At the end of 2 minutes (±30 seconds), the power failure flag (OFF) should disappear from view and the horizon line should indicate the attitude of the airplane. The pitch trim knob should be centered (reference mark in a horizontal position) when zero pitch is indicated.

i. After the power failure flag has disappeared, turn the pitch trim knob from its centered position to the extreme clockwise position. The sphere should move to indicate a dive attitude change of 10 to 15 degrees. Return the pitch trim knob to its centered position. Turn the pitch trim knob from its centered position to the extreme counterclockwise position. The sphere should move to indicate a climb attitude change of 5 to 10 degrees. Return the pitch trim knob to its centered position.

Note

Proper response to this movement indicates that the system is operating properly in the pitch axis. No similar check can be made on the roll axis, but the sphere should indicate the correct attitude.

j. Remove source of external power. The power warning flag (OFF) should appear at the upper left corner of the indicator.

6-56. VERTICAL GYRO.

6-57. The vertical gyro (figure 6-22) establishes the vertical reference line from which roll or pitch deviations of the airplane are measured. This vertical reference line is the spin axis of the gyro rotor assembly. The gyro housing, mounted within a gimbal assembly, is free to rotate on ball bearings about the pitch axis to a limit of 85 degrees in climb or dive; the gimbal assembly is mounted so that it is free to rotate about the roll axis throughout 360 degrees. Erection torque motors are provided in both the pitch axis and the roll axis to maintain the spin axis of the gyro perpendicular to the earth's surface at all times. Each erection torque motor can apply a force in either direction to precess the gyro back to its vertical or erect position. The indicating circuit, which is completely separate from the erection circuit of the gyro, consists of a position transmitter in the roll axis and another in the pitch axis. The transmitters measure the deviation of the airplane from level flight about each axis and transmit the information to the indicator.

6-58. VERTICAL GYRO CONTROL ASSEMBLY.

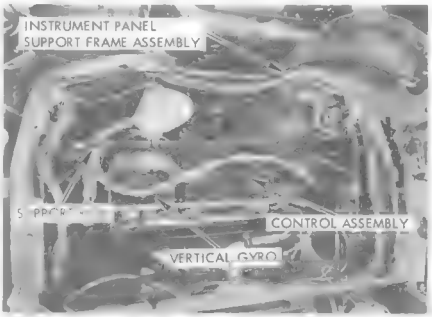
6-59. The vertical gyro control assembly (figure 6-22) houses the system's power supply assembly, the rate gyro, the time delay assembly, the relay box assembly and the roll and pitch amplifier channels. The control assembly receives 28-volt d-c and 115-volt a-c power directly from the primary d-c and a-c busses. The power supply assembly then furnishes the correct voltage and phase to all of the electrical circuits in the system. The rate gyro is provided to sense the direction and rate of turn and to operate the time delay assembly which, in turn, actuates the relays controlling the pitch-bank erection system in the vertical gyro. The time delay relay circuit, which delays action approximately 5 seconds, prevents the operation of the pitch-bank erection circuit during brief deviations in yaw, rough weather or minor changes in course. The amplifier channel in each attitude indicating system receives the voltage from the receiver synchro as a result of displacement at the vertical gyro and feeds the properly amplified voltage to the one-phase winding of the two-phase servomotor.

6-60. REMOVING AND INSTALLING VERTICAL GYRO AND VERTICAL GYRO CONTROL ASSEMBLY.

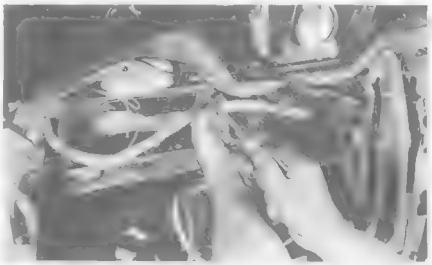
REMOVING

Note The vertical gyro only may be removed from the mounting tray without disturbing the control assembly. However, if the control assembly is to be removed, prior removal of the vertical gyro is necessary.

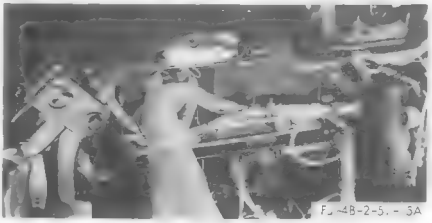
1 Remove instrument panel. (Refer to paragraph 6-19.)



2 Remove clamps securing wire bundle to support rod.



3 Remove connector from vertical gyro.

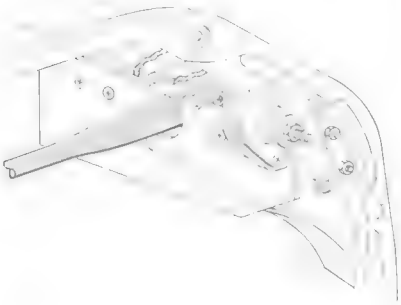


4 Cut safety wire from wing nuts on both units.

5 Loosen wing nuts to disengage securing bracket from mounting flange of each unit.

6 Turn right side of control assembly slightly aft and remove two connectors from right forward side of control assembly.

7 Remove two screws, nuts, washers, and spacer from right end of support rod.

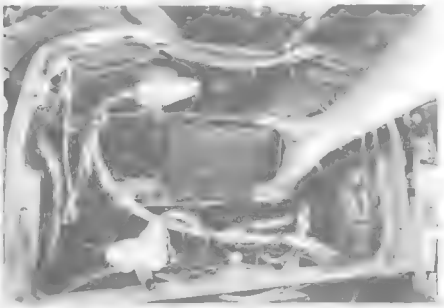


8 Spring rod upward, push vertical gyro down on aft end of mounting tray and pull under rod. Remove vertical gyro from airplane.

9 Slide control assembly toward center on mounting tray.

FJ-4B-2-51-16

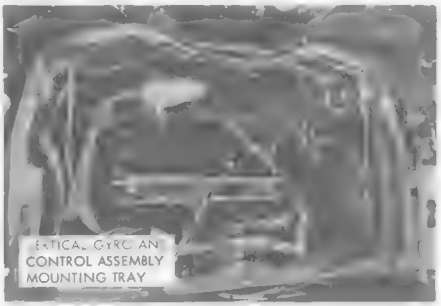
10 Spring rod upward, push control assembly down on aft end of mounting tray and pull aft under rod; remove control assembly from airplane.



INSTALLING

1 Spring support rod upward and slide control assembly under rod onto center of mounting tray.

2 Slide control assembly to right side of mounting tray.



3 Turn right side of control assembly aft to support rod for access to electrical connections; make two electrical connections.

4 Seat control assembly correctly on mounting tray and tighten wing nut to secure bracket against mounting flange of control assembly.

5 Spring support rod upward and slide vertical gyro under rod onto mounting tray.

6 Seat vertical gyro correctly on mounting tray and tighten wing nut to secure bracket against mounting flange of vertical gyro.

7 Safety-wire wing nuts on both units.

8 Make electrical connection at vertical gyro unit.

9 Secure support rod at right end with spacer, two screws, nuts and washers.

FJ-4B-2-51-17

10 Install clamps securing wire bundle to support rod.
11 Install instrument panel. (Refer to paragraph 6-21.)
12 Perform operational check of vertical gyro system. (Refer to paragraph 6-54.)

FJ-4B-2-51-18A

6-61. REMOTE ATTITUDE INDICATOR.

6-62. The remote attitude indicator, located on the instrument panel, provides the pilot with a constant visual indication of the simultaneous pitch and roll attitude of the airplane, relative to the earth's surface. The indicator consists of an indicating mechanism, pitch and roll servomotor assemblies, a power failure warning flag and a pitch trim adjustment knob. The indicating mechanism consists of a sphere assembly mounted on the indicator end plate. The indicating mechanism may be rotated a full 360 degrees about the horizontal axis and 85 degrees about the vertical axis. The output gear of the roll servo, mounted on the end plate, meshes with a gear on the sphere assembly, driving the assembly up or down to provide roll indications. The pitch servo drives the indicator sphere for pitch indications. A stationary miniature airplane, mounted on the indicator bezel, serves as the stable reference and is read against the moving sphere to interpret roll and pitch deviations. The power warning flag (OFF), located at the upper left corner of the indicator, comes into view indicating loss of a-c and/or d-c power to the system. A two-phase motor raises the flag from view when power is applied. Refer to paragraphs 6-7 and 6-8 for procedures on removing and installing attitude indicator.

6-63. INTERLOCK RELAY.

6-64. An a-c—d-c power interlock relay (figure 6-22) is installed in airplanes 143493k and subsequent. The relay is located forward of the instrument panel on the underside of the support assembly on which the vertical gyro and vertical gyro control assembly are mounted. The purpose of the relay is to prevent electrolytic switch wear and failure resulting from extended periods of high erection. This is accomplished by the simultaneous application of d-c power and three-phase a-c power to the vertical gyro system and the interruption of all power to the system in the event of a partial power loss.

6-64A. FUNCTION OF THE INTERLOCK RELAY—APPLYING POWER TO THE SYSTEM. The function of the interlock relay which ensures simultaneous application of d-c power and three-phase a-c power is described as follows (figure 6-23):

a. A-C power, phase "A" and phase "C," enters the relay at terminals 6 and 5, respectively. Phase "A" is routed through a full wave rectifier to the coil of a-c sensing relay R1. Phase "C" is routed through another full wave rectifier to the coil of a-c sensing relay R2. The resulting differences of potential energize relays R1 and R2.

b. When relays R1 and R2 become energized, d-c power (which has entered the relay at terminal 7) flows through the closed contacts of R1 and R2 to the coil of a-c/d-c control relay R3, energizing the relay (R3).

c. As relay R3 becomes energized, a-c power flows through the relay (R3) to another a-c/d-c control relay (R4) and d-c power flows through the relay (R3) to the coil of relay R4, energizing relay R4.

d. Upon the energizing of relay R4, d-c power and a-c power phase "A" and phase "C" are enabled to flow through relay R4 to the output terminals of the interlock relay.

6-64B. FUNCTION OF THE INTERLOCK RELAY—D-C POWER FAILURE. The interlock relay will interrupt all a-c power to the vertical gyro system in the event a d-c power failure is experienced to the interlock relay. This is done in the following manner (figure 6-23):

a. The absence of a d-c power input to the interlock relay will cause a-c/d-c control relays R3 and R4 to become de-energized, thus interrupting the flow of a-c power phase "A" and phase "C."

6-64C. FUNCTION OF THE INTERLOCK RELAY—ALL OR PARTIAL A-C POWER FAILURE. In the event a malfunction occurs in either one of the a-c power circuits to the interlock relay, the relay will open the d-c circuit and the remaining a-c circuit and prevent power from reaching the vertical gyro system. The operation of the interlock relay under such conditions is as follows (figure 6-23):

a. The absence of an a-c power phase "A" input to the interlock relay will cause a-c sensing relay R1 to become de-energized.

Note

The failure of "C" phase will be typical except that a-c sensing relay R2 will become de-energized.

b. Relay R1 or R2 being de-energized will keep d-c power from reaching the coil of a-c/d-c control relay R3 and consequently relay R3 will become de-energized; thus, any a-c power that is remaining will be interrupted at this point.

c. D-C power is still available to a-c/d-c sensing relay R4, however, and through a holding circuit keeps relay R4 energized.

d. A d-c circuit is also still available to a-c/d-c sensing relay R3 and when the relay (R3) becomes de-energized, a circuit is completed through the relay (R3) and also through relay R4 to energize two time delay relays R6 and MTR7.

e. The circuit to the coil of relay MTR7 contains a capacitor which starts charging as the circuit to relay MTR7 is completed.

f. When the capacitor reaches its full charge [which takes 3.5 (±1.5) minutes], relay MTR7 becomes de-energized and a circuit is completed through relay MTR7 and relay R6 to the coil of relay R5.

g. When relay R5 becomes energized, the holding circuit to relay R4 is broken and relay R4 is de-energized. This breaks the circuit to the d-c output terminal of the interlock relay and thus all power to the vertical gyro system has been halted.

6-65. REMOVING AND INSTALLING INTERLOCK RELAY.

REMOVING

1. Remove pilot's instrument panel. (Refer to paragraph 6-19.)

2. Remove vertical gyro, vertical gyro control assembly and shock mount from the vertical gyro support assembly. (Refer to paragraph 6-60.)

3. Pull rubber terminal covers from electrical terminals and slide back along wires.

4. Disconnect electrical wiring from relay.

5. Remove four mounting bolts and nuts and pull relay from mounting shelf and out of airplane.

INSTALLING

1. Position interlock relay on bottom side of mounting shelf and install four mounting bolts and nuts.

2. Connect electrical wiring to relay. (See figure 10-60.)

3. Install rubber terminal covers over electrical terminals.

4. Install shock mount, vertical gyro control assembly and vertical gyro on vertical gyro support assembly. (Refer to paragraph 6-60.)

5. Install pilot's instrument panel. (Refer to paragraph 6-21.)

FJ-48-2-51-159

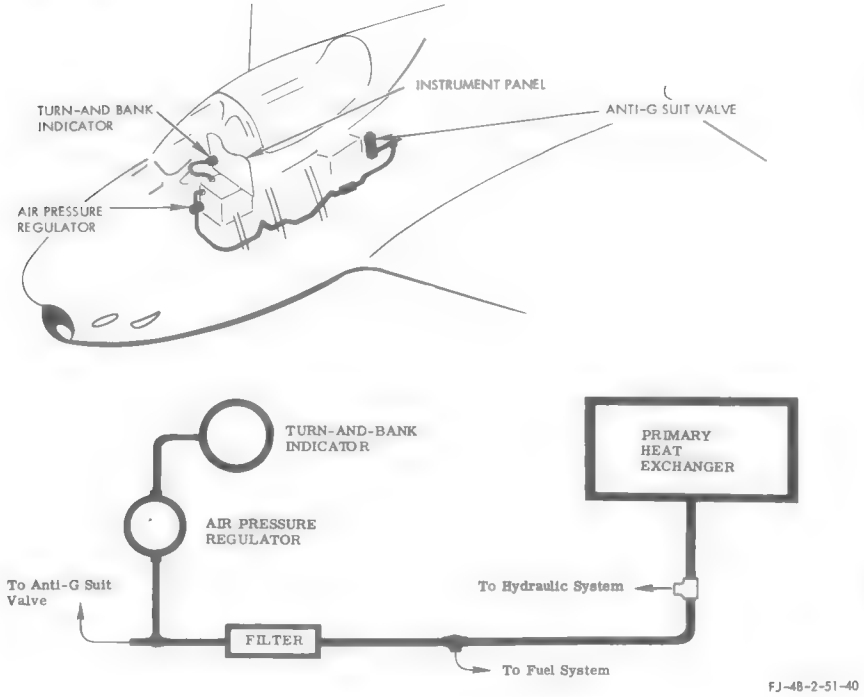


Figure No. 6-25. Turn-and-Bank Indicating System

TURN-AND-BANK INDICATING SYSTEM

6-66. TURN-AND-BANK INDICATING SYSTEM.

6-67. The turn-and-bank indicating system (figure 6-25) consists of a turn-and-bank indicator, an adjustable air pressure regulator and pressure lines from the heat and vent system. The primary heat exchanger furnishes thirteenth stage air to the heat and vent system from which air pressure is routed through a scale filter to the anti-G suit valve (in the anti-G suit system) and to the turn-and-bank air pressure regulator.

6-68. TROUBLE SHOOTING TURN-AND-BANK INDICATING SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
POINTER FAILS TO RESPOND.		
No air pressure supplied to the indicator.	Visually examine air line connection at the air pressure regulator on the center pedestal and at the turn-and-bank indicator.	Make proper connection.
	Check for clogged air line from center pedestal to turn-and-bank indicator.	Blow line clear.
	Check for clogged input or output port at air pressure regulator.	Clear port as required.
	Check all tubing and connections for leaks.	Repair or replace tubing; remake connections as necessary.
Dirt in indicator is restricting operation.		Replace indicator.
POINTER VIBRATES.		
Excessive vibration of indicator.	Check instrument mounting screws for tightness.	Tighten or replace as necessary.
	Check instrument panel stud fasteners for tightness.	Tighten studs as necessary.
Damping adjusting screw incorrectly adjusted.	Turn the damping adjusting screw slightly to the right to see if pointer vibration decreases.* (Refer to paragraph 6-71.)	Turn the damping adjusting screw to the right until sufficient damping is obtained.* (Refer to paragraph 6-71.)
Internal mechanism out of balance or defective.		Replace indicator.
POINTER DOES NOT SIT ON ZERO.		
Internal mechanism out of balance or defective.		Replace indicator.
Sensitivity spring has been adjusted incorrectly, pulling the pointer off zero.		Replace indicator.
POINTER FAILS TO RETURN TO ZERO.		
Defective or worn internal parts.		Replace indicator.

*Not to be accomplished if a replacement indicator is available. (Refer to paragraph 6-71.)

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
POINTER SLUGGISH IN RETURNING TO ZERO.		
Damping adjusting screw incorrectly adjusted.	Turn the damping adjusting screw slightly to the left to see if pointer moves more freely.* (Refer to paragraph 6-71.)	Turn the damping adjusting screw to the left until proper damping results.* (Refer to paragraph 6-71.)
Defective or dirty internal parts.		Replace indicator.
INCORRECT POINTER SENSITIVITY.		
Regulated air pressure too high or too low.	Check turn-and-bank air pressure regulator output. (Refer to paragraph 6-75.)	Adjust the outlet air pressure to 2 (±0.1) in. Hg.
Partially clogged air line from center pedestal to indicator.	Disconnect air line and check for dirt or obstruction.	Blow line clear.
Loose connection of air line to pressure regulator or to indicator.	Visually examine connections.	Make proper connections.
Sensitivity spring adjusted incorrectly.	During flight, it should be determined (when trouble is noted) if more or less pointer deflection is necessary for a standard rate turn.	Turn sensitivity adjusting screw out to allow the pointer to deflect further and turn screw in to allow the pointer to deflect less.* (Refer to paragraph 6-71.)
IN LOW TEMPERATURES, POINTER FAILS TO RESPOND OR DOES SO SLUGGISHLY AND WITH INSUFFICIENT DEFLECTION.		
Oil becomes too thick in indicator.		Replace indicator.
Defective internal mechanism.		Replace indicator.
GYRO FAILS TO START.		
No air pressure supplied to the indicator.	Refer to trouble POINTER FAILS TO RESPOND.	
Defective indicator.		Replace indicator.
NOISY GYRO.		
Defective indicator.		Replace indicator.

*Not to be accomplished if a replacement indicator is available. (Refer to paragraph 6-71.)

6-69. TURN-AND-BANK INDICATOR.

6-70. The turn-and-bank indicator, located on the instrument panel, is a direct reading instrument which provides an indication of the rate and co-ordination of turns. Indications provided by this instrument are used by the pilot for reference in keeping the airplane level laterally, in maintaining straight flight and in establishing the proper angle of bank for any rate of turn. The turn indicator is a rate instrument which consists of an air-driven gyro linked to a pointer. When the pointer is off-center, it indicates that the airplane is turning in the direction shown by the pointer. The amount that the pointer is off-center is proportional to the rate of turn. The instrument is a 4-minute turn indicator with two pointer widths (5/16 of an inch) deflection for 180 degrees per minute turn. The bank indicator is a ball-type inclinometer. Fluid in the curved inclinometer tube serves to damp out vibrations between the ball and the tube and to ensure smooth, accurate movement of the ball. The inclinometer tube is so constructed that when the airplane is flying level, the ball will be centered by

its own weight. When the airplane is making a turn, the ball is acted upon by gravity and centrifugal force. During a co-ordinated turn, both of these forces are such that the resultant force holds the ball centered in the tube. Thus, the centered ball indicates the proper lateral attitude for any rate of turn as well as for straight flight.

6-71. ADJUSTING TURN-AND-BANK INDICATOR.

Note

No attempt should be made to adjust the turn-and-bank indicator unless a replacement indicator is not available.

There are two adjusting screws on the turn-and-bank indicator located within the instrument case. When the damping adjustment screw is turned clockwise, damping is increased and when the screw is turned counterclockwise, damping is decreased. This adjusting screw is used if the turn pointer is vibrating or if the pointer is sluggish in returning to the zero position. When the sensitivity adjusting screw is turned counterclockwise, it allows the turn pointer to deflect further for a given rate of turn, and when the screw is turned clockwise, it reduces the deflection of the pointer. This adjustment is used if, during flight, it is determined that a co-ordinated turn of 180 degrees cannot be completed in one minute with two pointer widths (5/16 of an inch) deflection. If adjusting the sensitivity of the pointer pulls the pointer from its zero position, the indicator must be replaced.

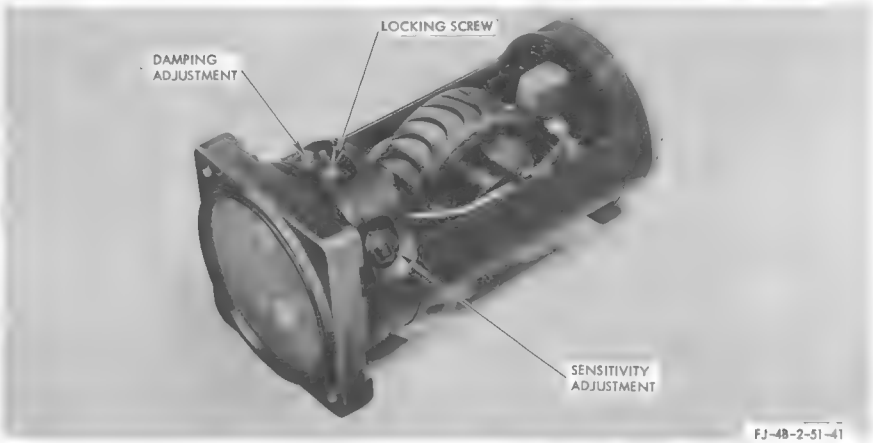


Figure No. 6-26. Turn-and-Bank Indicator Adjustment Screws

Figure No. 6-26A. Turn-and-Bank Air Pressure Regulator

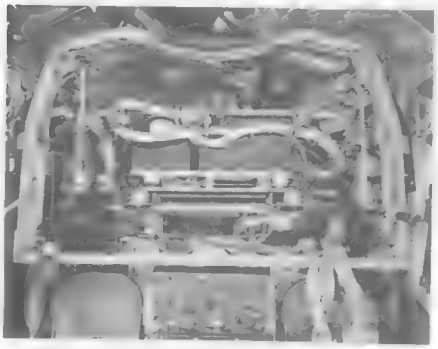
6-72. TURN-AND-BANK AIR PRESSURE REGULATOR.

6-73. An air pressure regulator is provided to reduce engine compressor pressure to the correct pressure differential above cockpit pressure for the operation of the turn-and-bank indicator. The regulator is located on the forward end of the pedestal assembly. Controlled pressure is adjustable from 1.0 in. Hg to 6.0 in. Hg, but should be set and maintained at 2.0 (±0.1) in. Hg for the correct rotating speed of the rate gyro in the indicator.

6-74. REMOVING AND INSTALLING TURN-AND-BANK AIR PRESSURE REGULATOR.

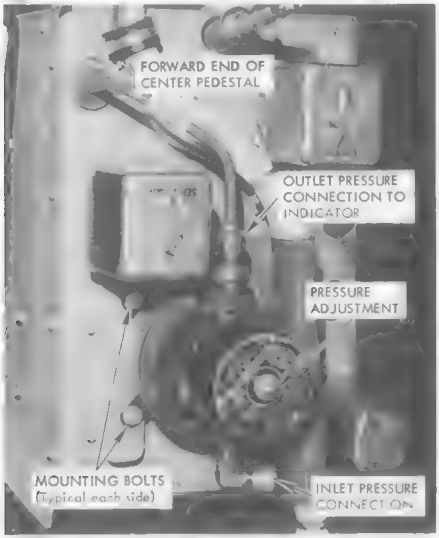
REMOVING

1 Remove instrument panel. (Refer to paragraph 6-19.)



2 With head and shoulders protruding through instrument panel frame, lean over center pedestal to reach pressure regulator.

3 Remove inlet and outlet pressure connections.



4 Remove four mounting bolts from mounting plate.

5 Clear the regulator from surrounding installed units, and lift the regulator from the airplane.

INSTALLING

1 Position the regulator and secure with four mounting bolts.

2 Connect inlet and outlet pressure lines to the proper fittings.

3 Check air pressure regulator output and adjust if necessary. (Refer to paragraph 6-75.)

4 Install instrument panel. (Refer to paragraph 6-21.)

FJ-48-2-51-42A

6-75. CHECKING AND ADJUSTING TURN-AND-BANK AIR PRESSURE REGULATOR OUTPUT. The air pressure regulator should be checked for output if it is suspected that the speed of the rate gyro is too fast or too slow. To check the pressure output, proceed as follows:

a. Remove the turn-and-bank indicator (paragraph 6-7) and pull the disconnected turn-and-bank air line through the cutout in the panel.

b. Connect a manometer and the turn-and-bank air line to the pressure port of the turn-and-bank indicator by using a "T" connector.

c. Detach the hose assembly from the inlet port of the anti-G suit filter in the left-hand forward radio bay. (Refer to paragraph 4-150.)

d. Attach a source of pressure and apply 35 psi pressure to the input line of the turn-and-bank air pressure regulator.

e. With 35 psi pressure applied to the inlet port of the regulator, check for an outlet pressure of 2 (±0.1) in. Hg.

f. If the outlet pressure is not 2 (±0.1) in. Hg, remove the "T" connector from the turn-and-bank indicator; then, remove the instrument panel. (Refer to paragraph 6-19.) Reinstall the "T" connector, the air line and the manometer to the turn-and-bank indicator.

g. With 35 psi pressure applied to the air pressure regulator, turn the pressure adjustment on top of the regulator clockwise to increase pressure and counterclockwise to decrease pressure until output is adjusted to 2 (±0.1) in. Hg pressure.

h. Vary the inlet pressure from 15 to 75 psi. The outlet pressure should not vary more than ±0.1 in. Hg.

CAUTION

Be sure that the ports of the regulator and the air line to the indicator are not clogged or sealed.

i. Remove the source of pressure and reconnect the hose assembly to the inlet port of the anti-G suit filter. (Refer to paragraph 4-150.)

j. Remove the manometer and the "T" connection from the turn-and-bank indicator.

k. Install the instrument panel. (Refer to paragraph 6-21.)

l. Install the turn-and-bank indicator. (Refer to paragraph 6-8.)

PITOT-STATIC SYSTEM

6-76. PITOT-STATIC SYSTEM.

6-77. The pitot-static system (figure 6-27) furnishes impact (pitot) pressure and atmospheric (static) pressures to the instruments and other equipment requiring these pressures for operating power. The system consists of an electrically heated pitot-static head, a pitot-static boom, five pitot line drain sumps, five static line drain sumps and tubing leading to the instruments. Although the pitot-static system is always considered as a single system, it is actually composed of two distinct systems: the pitot pressure system and the static pressure system; these two pressure systems *never* merge.

6-78. PITOT-STATIC PRESSURE LINES.

6-79. Pitot and static pressures are routed through rigid tubing from the boom assembly on the outer right wing panel along the front spar to the wing fold. Flexible lines are installed at the wing fold to permit folding the wing without damage to tubing or loss of pressure. Rigid tubing continues along the front spar of the wing and along the right side of the fuselage to the center pedestal. Pitot pressure is transmitted from the pedestal assembly through flexible lines to the airspeed and Mach number indicator. Static pressure is routed from the pedestal assembly to the airspeed and Mach number indicator and is then transmitted through interconnecting flexible lines to the sensitive altimeter and to the rate-of-climb indicator. A drain hole is provided in the pitot-static head and sumps are provided in the low sections of the lines to collect and drain moisture. To facilitate proper connection of lines to instruments, the fittings on the rear of the airspeed and Mach number indicator are marked with "P" (pitot) and "S" (static).

CAUTION

Always keep pitot and static openings, lines, fittings and instrument ports free of dirt and debris to ensure proper functioning of the instruments. *Cover the pitot-static head* when the airplane is on the ground and mask disconnected lines and instrument ports.

6-80. CHECKING PITOT PRESSURE LINE FOR LEAKS. To check the pitot pressure line for leaks, proceed as follows:

a. Make sure all pitot pressure lines are properly connected to their respective fittings and to all the instruments and equipment in the system. Make sure lines are not crossed.

b. Connect a source of pressure to the end of the pitot-static head. Seal drain hole with masking tape.

CAUTION

Do *not* apply suction to the pitot pressure line.

c. Apply pressure slowly until airspeed pointer indicates three-fourths of full range; then, pinch off source of pressure and secure with a suitable clamp.

d. The airspeed indication must not drop more than 2 knots after one minute.

Note

Energize the instrument panel vibrator, or tap the indicator, to keep pointer free from friction.

e. If airspeed indication drops more than 2 knots, a line leak, or internal leak in component, is indicated. To locate leak, perform isolation procedures. (Refer to paragraph 6-82.)

f. When there is no leak indicated, remove source of pressure from pitot-static head and remove masking tape from drain hole.

6-81. CHECKING STATIC PRESSURE LINE FOR LEAKS. To check the static pressure line for leaks, proceed as follows:

a. Make sure all static pressure lines are properly connected to their respective fittings and to all the instruments and equipment in the system. Make sure lines are not crossed.

b. Seal static pressure vents on pitot-static head.

c. Attach a source of suction to a static line drain sump. (See figure 6-27.)

CAUTION

- *Never* apply pressure to static pressure lines with instruments connected.
- Regulate the application and removal of suction by means of the indication of the rate-of-climb indicator which should not exceed 5000 feet per minute.

d. Set altimeter pointers to "0."

e. Slowly apply suction until altimeter reads 7000 feet; pinch off tube and secure with clamp.

CAUTION

To prevent damaging the pressure ratio transducer (Mach sensing trim system), do not exceed 7000 feet.

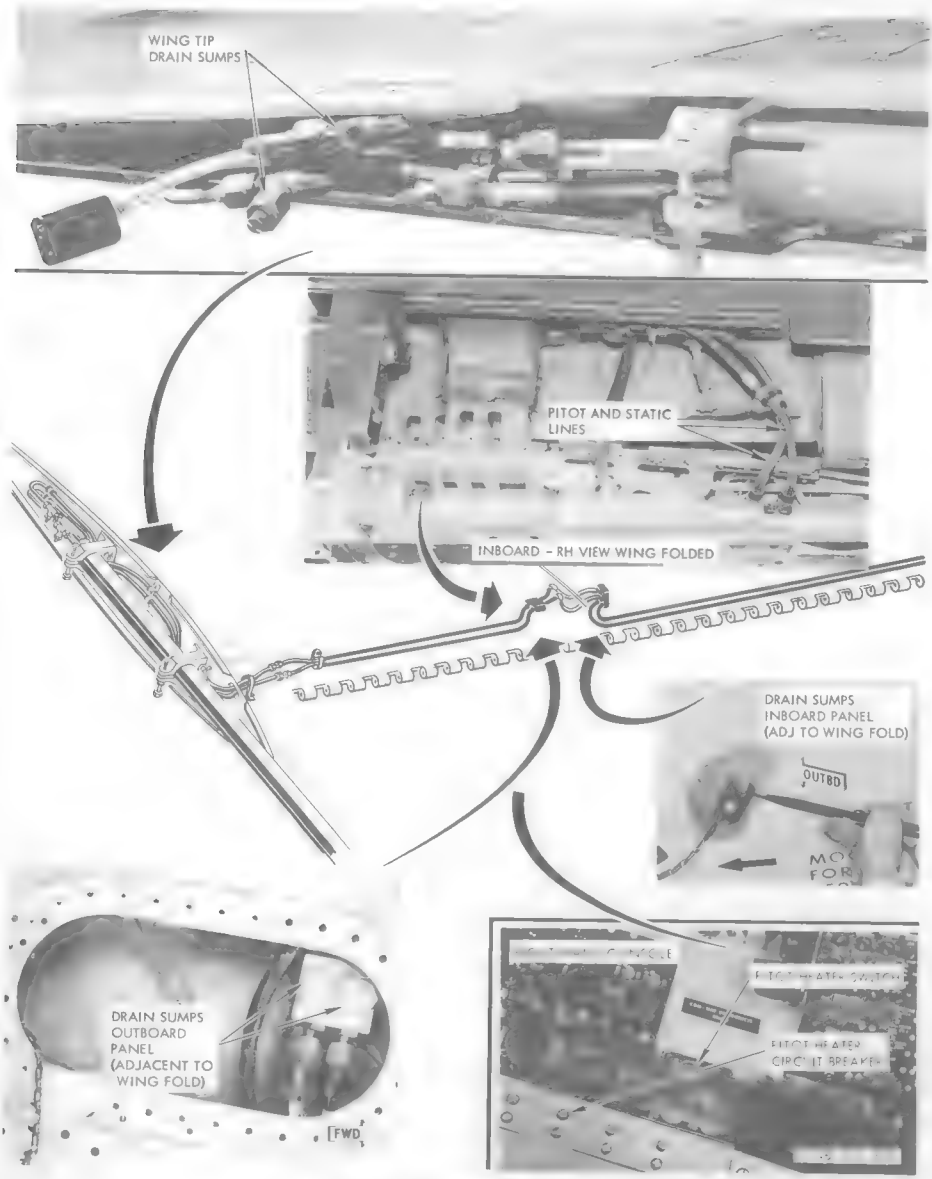


Figure No. 6-27. Pitot-Static System (Sheet 1)

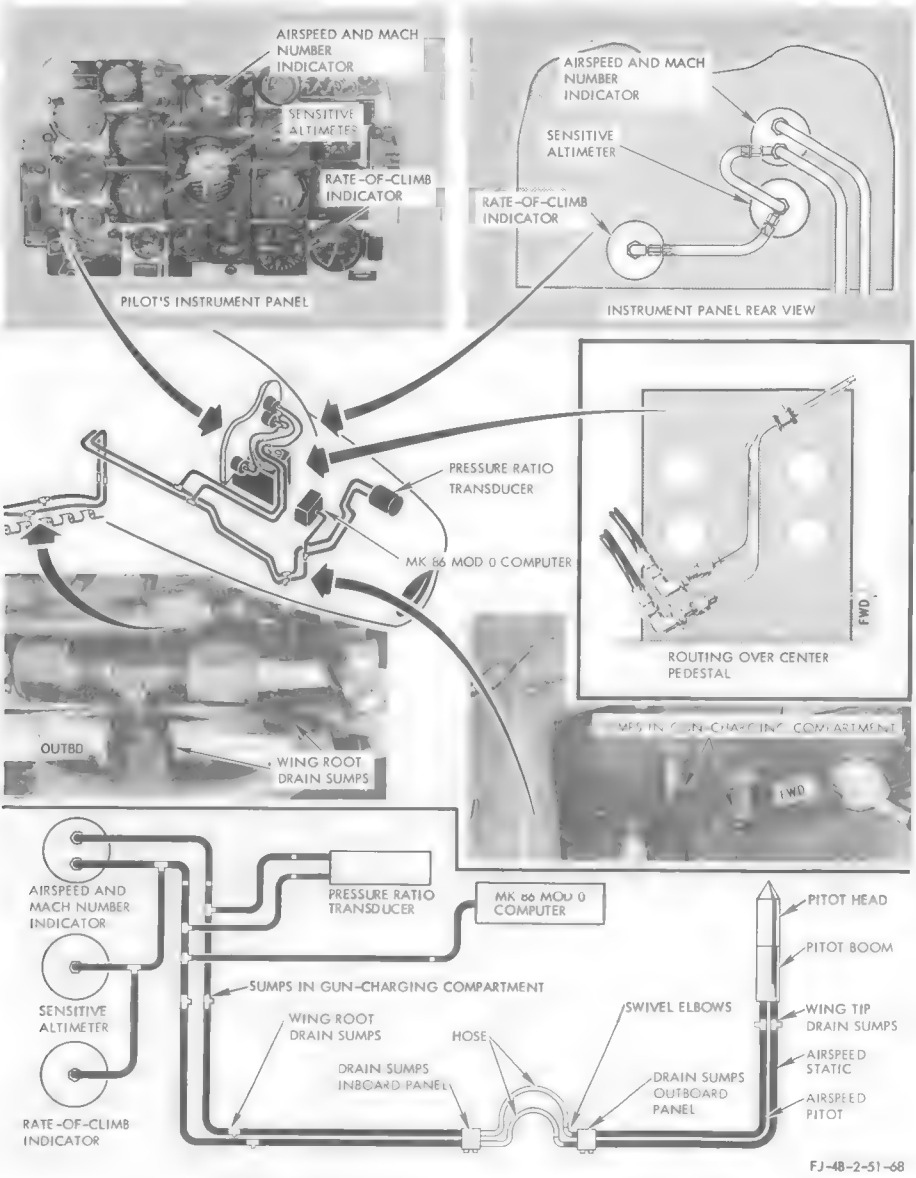


Figure No. 6-27. Pitot-Static System (Sheet 2)

f. Altimeter must not indicate less than 5000 feet after one minute. Energize instrument panel vibrator or tap altimeter to keep pointers moving freely.

g. If altimeter indicates less than 5000 feet, a line leak or component leak is indicated. To locate leak, perform isolation procedures. (Refer to paragraph 6-82.)

h. When there is no line leak indicated, slowly remove source of suction from static line drain sump and remove masking tape from static pressure vents on pitot-static head.

6-82. ISOLATING PITOT OR STATIC PRESSURE LINE LEAKAGE. If a line leak is indicated, proceed as follows:

a. Rotate fasteners on instrument panel counterclockwise and allow instrument panel to come aft to the position where safety cords are taut.

b. Disconnect pressure line to be checked.

c. Remove a section of flexible hose or tubing at fitting and plug all but one end.

d. Apply an air pressure of 10 in. Hg for one minute to the line. After a one-minute waiting period, the pressure should not drop below 9.95 in. Hg.

e. Repeat this procedure, each time removing another section of tubing, until leak is located.

f. Repair or replace damaged tubing.

g. Reconnect all lines in system and reconnect lines to the instruments.

h. Test lines for leaks. (Refer to paragraphs 6-80 and 6-81.)

i. Return instrument panel to its mounted position and secure by rotating fasteners clockwise.

6-83. CLEANING AND DRAINING PITOT OR STATIC PRESSURE LINES. To clean or drain the pitot or static pressure lines, proceed as follows:

a. Rotate the seven fasteners on instrument panel counterclockwise and allow instrument panel to move aft to the position where safety cords are taut.

b. Disconnect pressure line to be cleaned or drained.

c. Move all disconnected lines out of line of the fittings on the instruments.

CAUTION

Disconnected lines must be moved out of line of the instrument fittings to avoid having air or dirt blown into the instruments.

d. Open the five moisture drain sumps in the particular line which is clogged or which needs to be drained.

e. Using an R88-T-0903-020 (or more recent) test set, connect pressure outlet to one of the open moisture drain sumps. Apply pressure up to 10 psi until reasonably certain all moisture and dirt have been driven from line.

Note

Make certain the test set is delivering dry air in order to remove all moisture.

f. Remove test set and install plugs in all of the moisture drain sumps.

g. Connect lines to instruments.

h. Test the line for leaks. (Refer to paragraphs 6-80 and 6-81.)

i. Return the instrument panel to its mounted position and secure by rotating fasteners clockwise.

6-84. INSTALLING PITOT-STATIC SYSTEM FLEXIBLE LINES. To install pitot-static system flexible lines, proceed as follows:

a. Inspect lines for proper identification markings.

b. Blow flexible line clear with dry, high-pressure air.

c. Lubricate threads of male fittings sparingly with anti-seize compound (item 9, materials list).

CAUTION

Do not leave compound on the end of a fitting where it can enter the pitot or static system and cause improper functioning of the instruments.

d. Start nuts on fittings and turn nuts until finger-tight. Do not use a wrench until nuts are finger-tight.

e. With a torque wrench, tighten one nut to a torque of 70 inch-pounds (minimum torque). If connection leaks, tighten nut to a torque of 120 inch-pounds (maximum torque).

Note

Use a backup wrench on the flat of fitting so fitting will not turn when nut is tightened.

f. Check flexible line to be sure it is not twisted.

CAUTION

Prevent damaged lines or loose connections. If flexible line becomes twisted when it is installed, vibration and pressure will tend to untwist the line, loosen the fittings or shear off line ends.

g. Tighten nut on other end of line in same manner.

6-85. INSTALLING PITOT-STATIC SYSTEM RIGID TUBING. To install pitot-static system rigid tubing, proceed as follows:

a. Inspect tubing for proper identification markings.

b. Inspect tube flares for cracks, burrs, sharp edges and for concentricity with sleeves. Tight sleeves should not be considered a cause for rejection. Nuts, however, must turn freely on the sleeves.

c. Blow lines clean with dry, high-pressure air.

CAUTION

Dirt and metal chips will damage instruments. Open ends of tubing and fittings must be capped until they are to be connected in the system. When lines are disconnected from fittings, both lines and fittings should be capped or masked at once.

d. Lubricate threads of male fittings sparingly with anti-seize compound (item 9, materials list).

CAUTION

Do not leave compound on the end of a fitting where it can enter the pitot or static system and cause improper functioning of the instruments.

e. Place tube in position, making certain it has not been scratched, and ensure that tube flares meet fittings squarely and fully. *Never* use nut to draw flare to fitting as flare might be easily spun off or damaged.

f. Start nuts on fittings and turn until flares and sleeves are firmly seated. Do not use a wrench until nut is finger-tight.

g. After nut is finger-tight, use a torque wrench to tighten nut to 40 inch-pounds (minimum torque).

Note

Use a backup wrench on the flat of fitting so that fitting will not turn when nut is tightened.

h. If fitting leaks, nut may be tightened to 65 inch-pounds (maximum torque). If fitting still leaks, replace tubing or fitting as necessary.

6-86. PITOT-STATIC HEAD AND BOOM.

6-87. The straight-type pitot-static head is installed in the forward end of the pitot-static boom which is attached to the outboard rib of the right-hand outboard wing. The forward portion of the pitot-static head is open to receive the full force of impact pressure. The head is perforated on both the top and bottom to provide static pressure. A drain hole is provided to eliminate moisture. A resistance-type heater, installed in the pitot-static head, prevents formation of ice at the openings.

6-88. REMOVING PITOT-STATIC HEAD AND BOOM.

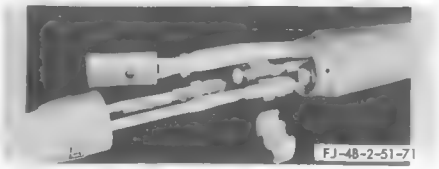
REMOVING HEAD

1. Remove protective cover from pitot-static head.
2. Remove four Nylok screws securing pitot-static head to boom sleeve.
3. Remove four screws and lock washers securing boom sleeve to boom.

4. Slide boom sleeve aft far enough to expose pitot and static line connections.

5. Disconnect electrical plug from pitot-static head receptacle.
6. Disconnect pitot and static lines from the pitot-static head and remove head.

Caution Do not twist lines.



7 Plug disconnected lines to prevent entry of dirt or moisture. Tie electrical plug in bag and cover receptacle with masking tape.

REMOVING BOOM

Note The pitot-static boom may be removed without prior removal of the pitot-static head.

8 Remove access panel on wing tip aft lower surface.

9 Disconnect pitot and static lines and electrical disconnect at rear of boom installation.

Caution Do not twist lines.

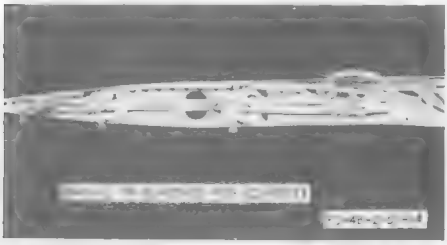
10 Loosen two clamps securing pitot-static boom.

11 Remove sealing compound from junction of boom and wing tip leading edge.

Caution Use wood or plastic scraper to avoid scratching wing surfaces.

12 Carefully pull boom forward out of both support clamps and free of the wing.

Caution Exercise care in removing the boom to avoid damaging the pitot or static lines, the electrical cable or wiring to the wing tip light.

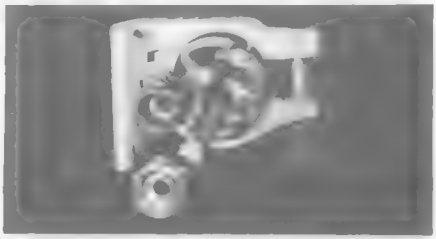


6-89. INSTALLING PITOT-STATIC HEAD AND BOOM.

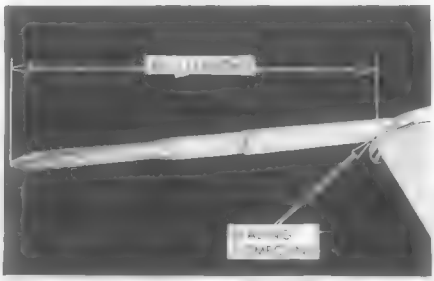
1 Insert boom in opening on wing leading edge and into forward support clamp attached to outer rib of outboard wing.

2 Slide boom aft through forward support clamp to aft support clamp.

3 Align slot in boom with pin in aft clamp; then slide boom aft until pin is bottomed in slot.



Caution The accuracy of airspeed indication is affected by length of the boom. The tip of the pitot-static head must be 25 (± 1) inches from the intersection of the wing leading edge and the centerline of the boom.



4 On wing tip lower surface, tighten boom securing screws at the two support clamp locations.

FJ-48-2-51-72

5 Make pitot and static line connections at "T" fittings aft of boom installation. (Refer to paragraph 6-85.)

6 Make electrical connection at aft end of boom.

7 Replace access panel.

8 Apply sealing compound (item 110, materials list) to junction of boom and wing tip leading edge.

9 Slide boom sleeve aft, remove plugs from lines and remove bag and masking tape from electrical connections.

10 Hold pitot-static head in position and connect pitot and static lines to fittings on pitot-static head. (Refer to paragraph 6-85 for correct procedure on installing pitot-static lines.)

Caution Do not blow lines clear with instruments connected. Do not twist lines.

Note Make certain the pitot-static head is installed with the word "TOP" on the upper surface and the drain holes at the bottom.

11 Connect electrical plug to pitot-static head receptacle.

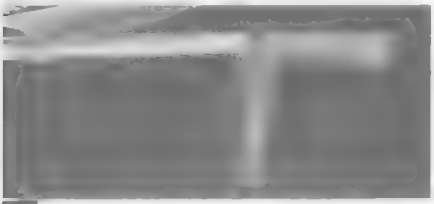
12 Pull boom sleeve forward and secure with four screws and lock washers.

13 Paint both contact surfaces between pitot-static head and boom sleeve with zinc chromate primer (item 102, materials list); push head into forward end of boom sleeve and install four Nylok screws while primer is still wet.

Caution Keep paint off electrical connection.

14 Check pitot and static pressure lines for leaks. (Refer to paragraphs 6-80 and 6-81.)

15 Check pitot heater circuit operation. (Refer to paragraph 6-96.)



16 Place protective cover over pitot-static head.

FJ-48-2-51-73

6-90. CLEANING PITOT TUBE OPENING. The forward end of the pitot-static head should be kept free of dirt at all times to admit the full force of impact pressure. To clean the tube opening, use a soft brush or a cloth dampened with water. If necessary, remove the pitot-static head (paragraph 6-88) and attach a source of air pressure to the aft end to expel any particles of dirt or debris.

6-91. CLEANING STATIC VENTS. The static pressure vents should be cleaned in the same manner as the pitot tube opening. (Refer to paragraph 6-90.)

6-92. CLEANING PITOT-STATIC HEAD AND BOOM. To clean the exterior surface of the pitot-static head and boom, refer to paragraph 1-59.

6-93. PITOT HEATER.

6-94. A resistance-type, anti-icing heater unit is an integral part of the pitot-static head. The heating unit will not burn out under normal operating conditions. A switch (PITOT HEATER), located on the aft end of the right-hand console panel, controls the operation of the pitot heater circuit. (See figure 6-28.)

Note

When the airplane is on the landing gear, ground safety relay No. 2, controlled by the ground safety switch, will interrupt the pitot heater circuit. This protective feature prevents burning out the element in the event the PITOT HEATER switch is left in the "ON" position.

Power is supplied by the primary 28-volt d-c bus through a 15-ampere circuit breaker located on the right-hand console panel. (See figure 6-28.)

PRIMARY 28V DC
5A
GROUND SAFETY (1)
GROUND SAFETY & OVERVOLTAGE (2)
LH Radio Bay Circuit-breaker Panel

PRIMARY 28V DC
15A PITOT HEATER
Off
Pitot Heater
PITOT HEATER SWITCH
RH Forward Console Panel

PITOT HEATER
PITOT - STATIC BOOM
RH Wing Tip

D
E
B
A
GROUND SAFETY RELAY NO. 2

Loaded
Unloaded
GROUND SAFETY SWITCH
LH Main Gear Strut

A
B
GROUND SAFETY RELAY NO. 1
Canopy Deck Relay Panel

(1) Airplanes 139531 through 139555
(2) Airplanes 141444 and subsequent

FJ-4B-2-51-143A

Figure No. 6-28. Pitot Heater

6-95. TROUBLE SHOOTING PITOT HEATER.

TEST EQUIPMENT: D-C voltmeter.
Ohmmeter.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
PITOT HEATER and GROUND SAFETY* (GROUND SAFETY & OVERVOLTAGE†) circuit breakers engaged.
Airplane on jacks or ground safety switch linkage disconnected and moved to the unloaded position. (Refer to paragraph 6-96.)
PITOT HEATER switch in "ON" position.

CAUTION

Do not allow heater element to overheat.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
PITOT HEATER INOPERATIVE.			
Pitot heater element defective.	Check between test point FPA and ground.	28 volts dc.	Replace pitot-static head. (Refer to paragraphs 6-88 and 6-89.)
		Zero volts.	Continue trouble shooting.
Defective wiring or defective PITOT HEATER switch.	Check between test point FPB and ground.	28 volts dc.	Continue trouble shooting.
		Zero volts.	Replace defective PITOT HEATER switch or defective power wires.
Defective wiring or defective ground safety switch.	Check between test point GDE and ground.	Zero ohms.	Continue trouble shooting.
		Other than zero ohms.	Refer to paragraph 3-102, Trouble Shooting Landing Gear System.
Defective wiring or defective GROUND SAFETY RELAY NO. 2.	Check between test point GM and ground.	28 volts dc.	Replace defective GROUND SAFETY RELAY NO. 2 or defective wire(s) to pitot heater element disconnect.
		Zero volts.	Replace defective relay power wire.
D-C POWER FAILURE.			
Circuit-breaker failure.	Check between test points PGR and PBJ and ground.	28 volts dc.	Replace defective circuit breaker(s).
		Zero volts.	Refer to paragraph 8-61, Trouble Shooting D-C Power Distribution System.

*Airplanes 139531i through 139555i not having Service Change No. 374 complied with
†Airplanes 141444j and subsequent and airplanes having Service Change No. 374 complied with

6-96. CHECKING PITOT HEATER CIRCUIT OPERATION. To check the pitot heater circuit, proceed as follows:

a. Disable the ground safety switch on the left-hand landing gear to energize the ground safety relays. To disable the switch, remove cotter pin, washer and pin securing actuator link to switch arm. Install warning flag on switch arm. (See figure 1-14.)

WARNING

Make certain the STORES JETTISON & DROP TANK TRANSFER circuit breaker, located on the left-hand forward console panel, is pulled.

b. Depress PITOT HEATER circuit breaker on right-hand forward console panel. (See figure 6-28.)

c. Place the d-c power switch, located on the right-hand forward console panel, in the "OFF" position.

d. Connect a source of external power to the airplane.

e. Push the PITOT HEATER switch on the right-hand forward console panel to the "HEATER" position.

f. Feel pitot-static head for evidence of heat. If the pitot-static head becomes warm to the touch, turn PITOT HEATER switch to "OFF" position, pull circuit breaker, remove external power and reverse procedures performed in step a. If pitot-static head does not become warm to touch, perform pitot heater trouble shooting procedures. (Refer to paragraph 6-95.)

6-97. (Deleted.)

6-98. PITOT-STATIC OPERATED INSTRUMENTS.

6-99. Three diaphragm, pressure-operated flight instruments, located on the pilot's instrument panel, are connected directly to the pitot-static system. (See figure 6-27.) Both pitot and static pressures are furnished the airspeed and Mach number indicator, while static pressure only is supplied to the rate-of-climb indicator and the sensitive altimeter. Static pressure is supplied to the Mark 86 Mod 0 computer in the fire control system. Pitot and static pressures are also routed to the pressure ratio transducer (in the Mach sensing trim and angle-of-attack and angle-of-sideslip systems).

6-100. PREVENTIVE MAINTENANCE.

6-101. To ensure correct operation of the pitot-static operated instruments, the following checks should be performed:

a. Examine the pitot-static head and boom for correct installation. The boom should be secure and should be positioned so that the tip of the boom extends 25 (± 1) inches from the intersection of the wing leading edge and the centerline of the boom.

CAUTION

Airspeed indicator accuracy is greatly affected if the airspeed boom is not positioned correctly.

b. Examine the pitot-static head and boom for bent or damaged exterior surfaces, or for corrosion and dirt in or near pitot or static openings. Inaccurate indications will result if any of these conditions exist. Replace pitot-static head or boom if damaged. (Refer to paragraphs 6-88 and 6-89.) Thoroughly clean pitot or static tube openings if dirt or obstructions are found. (Refer to paragraphs 6-90 and 6-91.)

c. Examine pitot-static pressure connections to the instruments. Lines must be connected tightly to the correct fittings on the instruments. To gain access to connections, rotate the seven stud fasteners on front of instrument panel counterclockwise and allow panel to come aft to the position where safety cords are taut. After examining connections, restore panel to its mounted position and tighten fasteners.

6-102. RATE-OF-CLIMB INDICATOR.

6-103. The rate-of-climb indicator measures the rate at which an airplane in flight is changing altitude. The operation of the rate-of-climb indicator is based on changes in atmospheric pressure as a result of changes in altitude. Static pressure from the pitot-static tube connections enters a diaphragm within the case and from the same connection enters the interior of the case through a restrictor. The pressure change in the diaphragm is more rapid than the pressure change in the case so that a differential pressure is established. This pressure differential is transmitted to the dial by means of mechanical linkage. The pointer assembly indicates on a scale calibrated from 0 to 6, indicating climb or descent in thousands of feet per minute.

6-104. TROUBLE SHOOTING RATE-OF-CLIMB INDICATOR.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
POINTER FAILS TO RESPOND.		
Static pressure line clogged.	Disconnect the static line from all three flight instruments. Disconnect the static line from the pedestal assembly. Disconnect the static tubing if necessary. Remove drain plugs. (See figure 6-27.)	Blow dry, high-pressure air through the disconnected lines or tubing to remove dirt or obstruction. **CAUTION** Be certain that all equipment using static pressure has been disconnected before applying air pressure.
Defective indicator.		Replace indicator.
POINTER OFF "0" POSITION WHEN AIRPLANE IS NOT IN MOTION.		
Mechanism shift within the indicator.	Return the pointer to the "0" position by using the slotted zero adjusting shaft. Tap the indicator lightly when making the adjustment.	If pointer cannot be reset by means of the zero adjusting shaft, replace the indicator.
POINTER INDICATES INCORRECTLY.		
Partially kinked, twisted or clogged static line.	Visually inspect lines. Remake connections if line is kinked or twisted. If line is straightened and trouble persists, disconnect all static lines from instruments.	Replace badly bent or damaged lines. Blow lines clear with dry, high-pressure air. **CAUTION** Be certain that all equipment using static pressure has been disconnected before applying air pressure.
Pitot-static boom bent or damaged.	Visually inspect pitot-static boom.	Replace pitot-static boom if necessary. (Refer to paragraphs 6-88 and 6-89.)

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
POINTER INDICATES INCORRECTLY. (Cont)		
Leak in static line, tubing or in connections.	Check static pressure line for leaks. (Refer to paragraph 6-81.) Isolate line leakage. (Refer to paragraph 6-82.)	Eliminate leak. Repair or replace tubing as necessary.
Friction in indicator mechanism.		Replace indicator.
Leak in indicator case.		Replace indicator.
POINTER VIBRATES.		
Excessive vibration of instrument panel.	Check stabilizers.	Replace stabilizers if necessary.
Excessive vibration of static lines.	Visually inspect static connections and lines on rear of instrument panel.	Make proper, tight connections.
Defective indicator.		Replace indicator.
POINTER OSCILLATES.		
Loose mechanism parts within indicator.		Replace indicator.
Leak in static flexible lines or tubing.	Check static pressure lines for leaks. (Refer to paragraph 6-81.) Isolate line leakage. (Refer to paragraph 6-82.)	Eliminate leak. Repair or replace tubing as necessary.

6-105. ADJUSTING RATE-OF-CLIMB INDICATOR. A zero adjusting shaft with a slotted end (figure 6-29) is provided to set the indicating pointer on "0." When the altitude is not changing (when the pressure on the inside of the diaphragm equals the pressure within the case), the pointer should be settled on the "0" position. To adjust the pointer to "0," insert a screwdriver in the slotted end of the zero adjusting shaft and turns as much as necessary in the required direction.

Note

Operate the instrument panel vibrator or tap the instrument lightly while making the zero setting adjustment.

After adjusting the pointer position, seal the adjusting device by applying a 1/16-inch wide white marker across the slotted screw head.

6-106. SENSITIVE ALTIMETER — AIRPLANES NOT HAVING SERVICE CHANGE NO. 517 COMPLIED WITH.

6-107. The sensitive altimeter (figure 6-30) indicates the altitude corresponding to the existing pressure of the atmosphere surrounding the airplane. Three concentric pointers indicate feet of altitude on a common dial. A

Figure No. 6-29. Rate-of-Climb Indicator Zero Adjusting Shaft

large pointer indicates in units of 100 feet, a second pointer indicates in units of 1000 feet and a small pointer indicates in units of 10,000 feet. A barometric scale, visible through a window on the altimeter dial, is set to indicate pressure in inches of mercury. The altimeter has a range of 0 to 50,000 feet and the barometric scale can be set from 28.1 to 31.0 in. Hg. Two setting marks are located on the altimeter dial to indicate barometric pressure in feet of altitude when the barometric pressure setting is outside the limits of the indicated scale. Prior to flight, the existing barometric pressure is manually set on the scale by means of a knob at the lower left-hand corner of the case. (See figure 6-30.) Simultaneously, the pointers rotate to indicate field elevation and the index marks indicate barometric pressure in feet. Within the instrument case, the sensitive altimeter contains an assembly consisting of three evacuated diaphragms and a precision mechanism which multiplies diaphragm deflection. Static pressure from the pitot-static system is admitted to the altimeter case. As the diaphragms expand or contract with changes in case pressure, the motion is transmitted to the pointers and the scale.

6-107A. COUNTER-POINTER ALTIMETER — AIRPLANES HAVING SERVICE CHANGE NO. 517 COMPLIED WITH.

6-107B. The counter-pointer altimeter (figure 6-30) indicates the altitude corresponding to the existing pressure of the atmosphere surrounding the airplane. A sweep pointer indicates in units of 100 feet; a counter, located on the left-hand side of the altimeter, indicates in units of 1000 feet. Each time the sweep pointer completes one revolution, the counter will click over one number, thereby indicating the number of clockwise revolutions completed by the pointer. A barometric scale, visible through a window on the right-hand side of the altimeter dial, is set to indicate pressure in inches of mercury. The altimeter has a range of −1000 to 80,000 feet and the barometric scale indicates from 28.1 to 31.0 in. Hg. Prior to flight, the existing barometric pressure is manually set on the scale by means of a knob at the lower left-hand corner of the case (figure 6-30). The counter-pointer altimeter contains an assembly consisting of an evacuated diaphragm and a counter mechanism. Static pressure from the static system is admitted to the altimeter case. As the diaphragm contracts or expands with the changes of pressure on the case, the motion is transmitted to the sweep pointer, which in turn is conveyed by mechanical linkage to the counter mechanism.

6-108. TROUBLE SHOOTING THE SENSITIVE AND COUNTER-POINTER ALTIMETERS.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
POINTERS FAIL TO RESPOND.			
Static pressure line clogged.	Disconnect the static line from all three flight instruments. Disconnect the static line from the pedestal assembly. Disconnect static tubing as necessary. Remove drain plugs. (See figure 6-27.)		Blow dry, high-pressure air through the disconnected lines or tubing to remove dirt or obstruction. **CAUTION** Be certain that all equipment using static pressure has been disconnected before applying air pressure.
Defective indicator.			Replace indicator.
POINTERS INDICATE INCORRECTLY.			
Pitot-static boom bent or damaged.	Visually inspect pitot-static boom.		Replace pitot-static boom if necessary. (Refer to paragraphs 6-88 and 6-89.)
Pitot line connected to static port at airspeed and Mach number indicator.	Visually check pitot and static lines from pedestal assembly to instruments.		Reverse connection if necessary.
Leak in static pressure line.	Check static line for leaks. (Refer to paragraph 6-81.) Isolate line leakage. (Refer to paragraph 6-82.)		Eliminate leak. Repair or replace tubing as necessary.
Defective instrument.			Replace instrument.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
POINTER(S) VIBRATES.			
Excessive vibration of instrument panel.	Check stabilizers.		Replace stabilizers if necessary.
Excessive vibration of static lines.	Visually inspect static connections and lines on rear of instrument panel.		Make proper, tight connections.
Defective instrument.			Replace instrument.
POINTER(S) OSCILLATE UNDER VIBRATION.			
Defective instrument.			Replace instrument.
ERRATIC MOVEMENT OF POINTER(S).			
Defective instrument.			Replace instrument.
INCORRECT BAROMETRIC PRESSURE READING.			
Shock has shifted the instrument mechanism.	Check and adjust the barometric scale setting. (Refer to paragraph 6-111.)		Replace the instrument if the barometric scale can not be corrected.

TEST EQUIPMENT: D-C voltmeter.

SYSTEM CONDITIONS: 115-volt a-c external power.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
INTEGRAL VIBRATOR FAILS TO OPERATE — AIRPLANES HAVING SERVICE CHANGE NO. 517 COMPLIED WITH.			
Defective instrument.	Check test points FQA and FQB to ground.	115 volts ac.	Replace defective instrument.
		Zero volts.	Continue trouble shooting.
Power failure.	Check test point XCT to ground.	115 volts ac.	Replace defective fuse or wire segment.
		Zero volts.	Refer to paragraph 8-78, Trouble Shooting A-C Power Supply and Distribution System.

6-109. SENSITIVE AND COUNTER-POINTER ALTIMETER CASE LEAK TEST. To check for case leakage, proceed as follows:

a. Remove altimeter from the instrument panel. (Refer to paragraph 6-7.)

b. Disconnect static line from sensitive altimeter. Cover static line opening to keep out foreign matter.

c. Connect a source of suction to instrument and *slowly* lower pressure until altimeter indicates 18,000 feet; close off suction. Pointer should not change more than 100 feet in one minute.

d. Slowly remove vacuum and disconnect source of suction.

e. Attach static pressure line to instrument.

f. Install altimeter. (Refer to paragraph 6-8.)

6-110. SENSITIVE ALTIMETER FRICTION TEST — AIRPLANES NOT HAVING SERVICE CHANGE NO. 517 COMPLIED WITH. To check for friction within the sensitive altimeter case, proceed as follows:

a. Remove altimeter from instrument panel. (Refer to paragraph 6-7.)

b. Disconnect static line from sensitive altimeter. Cover static line opening to keep out foreign matter.

c. Connect a source of suction to instrument port and *slowly* lower pressure, without vibration, until altimeter indicates 10,000 feet.

d. Tap altimeter lightly; change in pointer reading should not exceed 120 feet.

e. Slowly remove vacuum and disconnect source of suction.

f. Attach static pressure line to indicator.

g. Install altimeter. (Refer to paragraph 6-8.)

6-110A. COUNTER-POINTER ALTIMETER FRICTION TEST — AIRPLANES HAVING SERVICE CHANGE NO. 517 COMPLIED WITH. To check for friction within the counter-pointer altimeter case, proceed as follows:

a. Remove altimeter from instrument panel. (Refer to paragraph 6-7.)

b. Disconnect static line from counter-pointer altimeter. Cover static line opening to keep out foreign matter.

c. Disconnect electrical disconnect from counter-pointer altimeter. (This operation stops the integral vibrator.)

d. Connect a source of suction to instrument port and *slowly* lower pressure until altimeter indicates 10,000 feet.

e. Slowly reduce the vacuum until the rate of descent indicates 250 feet per minute. As the sweep pointer reaches zero and the next number in the counter window clicks over, the change in sweep pointer reading should not exceed 300 feet.

f. Reconnect the electrical disconnect which starts the integral vibrator.

g. Repeat step d.

h. Slowly reduce the vacuum until the rate of descent indicates 250 feet per minute. As the sweep pointer reaches zero and the next number in the counter window clicks over, the change in the sweep pointer reading should not exceed 25 feet.

i. Slowly remove vacuum and disconnect source of suction.

j. Attach static pressure line to indicator.

k. Install altimeter. (Refer to paragraph 6-8.)

6-111. CHECKING AND ADJUSTING SENSITIVE AND COUNTER-POINTER ALTIMETER BAROMETRIC SCALE SETTINGS. To check the barometric scale setting of the sensitive altimeter, proceed as follows:

a. Rotate adjusting knob, located in the lower left-hand corner of instrument case, to set hands of altimeter to the surveyed field elevation.

Note

Operate instrument panel vibrator or tap the instrument lightly to remove all friction.

b. Compare barometric scale of airplane's altimeter with barometric scale of a portable test altimeter that has been adjusted at the control tower for correct altimeter setting. If barometric pressure scale of airplane's altimeter does not indicate the existing altimeter setting at field elevation, proceed to step c.

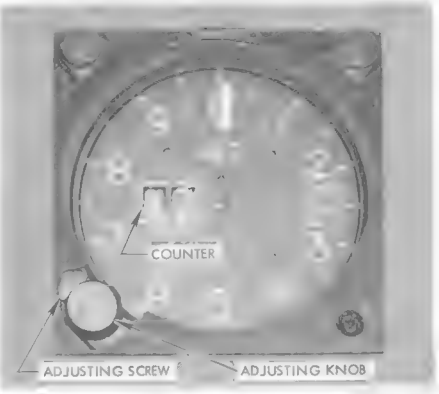
c. Loosen adjusting screw, found just to the left of the adjustment knob (figure 6-30), until head clears flange; then, move screw to the left. *Do not remove the screw.*

Note

Moving the adjusting screw out of position allows the adjustment knob, when pulled out, to turn only the barometric dial and setting marks.

d. With hands of altimeter still indicating correct field elevation, pull out setting knob and turn it until barometric scale indicates the same as the test altimeter. Re-check pointer and scale indications.

SENSITIVE ALTIMETER



COUNTER-POINTER ALTIMETER

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Figure 6-30. Sensitive and Counter-pointer Altimeter Adjustment Knobs and Adjusting Screws

e. Push knob in, being careful not to change the settings.

f. Return screw to its original position and tighten.

6-112. AIRSPEED AND MACH NUMBER INDICATOR.

6-113. The airspeed and Mach number indicator incorporates a conventional airspeed indicator and an altitude mechanism to provide a continuous indication of indicated airspeed and Mach number on the same instrument dial. On the face of the instrument is a fixed airspeed dial, graduated in knots, and a rotating dial, graduated

in Mach number. At low altitudes, part of the movable Mach dial is masked by the fixed airspeed dial. Both indicated airspeed and Mach are read from the same indicating pointer, but airspeed alone will be indicated below 150 knots. The range of the instrument is 80 to 650 knots indicated airspeed and from 0.5 to 2.0 Mach with a maximum operating limit of 50,000 feet altitude. The indicating pointer is actuated by the airspeed mechanism which is operated by a sensitive differential pressure diaphragm to which pitot pressure is admitted. The rotating dial is operated by a sensitive altimeter diaphragm mechanism to which static pressure is admitted. There is no interconnecting linkage between the two indicating mechanisms, but gearing between the moving scale and the altitude mechanism is such that Mach number will be indicated by the pointer at any combination of airspeed and altitude within the limits of the Mach number scale. Two setting indexes are provided, one for Mach number and one for indicated airspeed. The airspeed index is used for landing purposes and the Mach index is a Mach limit warning.

6-114. TROUBLE SHOOTING AIRSPEED AND MACH NUMBER INDICATOR.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
INDICATED AIRSPEED POINTER FAILS TO RESPOND.		
Pitot pressure line clogged.	Disconnect the pitot line from pedestal to indicator. Disconnect pitot tubing as necessary. Remove drain plugs. (See figure 6-27.)	Blow dry, high-pressure air through the disconnected line or tubing to remove dirt or obstruction. **CAUTION** Be certain all equipment using pitot pressure has been disconnected before applying air pressure.
Defective indicator.		Replace indicator.
MACH NUMBER ROTATING DIAL FAILS TO RESPOND.		
Static pressure line clogged.	Disconnect the static line from pedestal and from indicator. Disconnect static tubing as necessary. Remove drain plugs. (See figure 6-27.)	Blow dry, high-pressure air through disconnected line or tubing to remove dirt or obstruction. **CAUTION** Be certain all equipment using static pressure has been disconnected before applying air pressure.
INDICATED AIRSPEED POINTER INDICATES INCORRECTLY.		
Pitot-static boom bent or damaged.	Visually inspect pitot-static boom.	Replace pitot-static boom if necessary. (Refer to paragraphs 6-88 and 6-89.)
Leak in pitot pressure line or tubing.	Check all connections. Check pitot pressure line for leaks. (Refer to paragraph 6-80.) Isolate line leakage. (Refer to paragraph 6-82.)	Tighten connections. Eliminate leak. Repair or replace tubing as necessary.
Leak in indicator.	Perform diaphragm leak test (paragraph 6-118) and case leak test (paragraph 6-119).	Replace indicator.
MACH NUMBER ROTATING DIAL INDICATES INCORRECTLY.		
Leak in static pressure line or tubing.	Check all connections. Check static pressure line for leaks. (Refer to paragraph 6-81.) Isolate line leakage. (Refer to paragraph 6-82.)	Tighten connections. Eliminate leak. Repair or replace tubing as necessary.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
MACH NUMBER ROTATING DIAL INDICATES INCORRECTLY. (Cont)		
Leak in indicator.	Perform diaphragm leak test (paragraph 6-118) and case leak test (paragraph 6-119).	Replace indicator.
INDICATED AIRSPEED POINTER VIBRATES.		
Excessive vibration of instrument panel.	Check seven stud fasteners on front of instrument panel for tightness. Check stabilizers.	Tighten fasteners. Replace stabilizers if necessary.
Defective indicator.		Replace indicator.
INDICATED AIRSPEED POINTER OFF "0" WHEN AIRPLANE IS AT REST.		
Defective indicator.		Replace indicator.
INDICATED AIRSPEED POINTER OSCILLATES.		
Leak in pitot pressure line or tubing.	Check all connections. Check pitot pressure line for leaks. (Refer to paragraph 6-80.) Isolate line leakage. (Refer to paragraph 6-82.)	Tighten connections. Eliminate leak. Repair or replace tubing as necessary.
Leak in indicator.	Perform diaphragm leak test (paragraph 6-118) and case leak test (paragraph 6-119).	Replace indicator.
ERRATIC MOVEMENT OF INDICATED AIRSPEED POINTER OR MACH NUMBER ROTATING DIAL.		
Friction within indicator case.	Defective indicator.	Replace indicator.

6-115. SETTING AIRSPEED AND MACH NUMBER INDEXES. A setting knob (MACH PUSH LIMIT), located in the lower left-hand corner of the instrument case, is provided to manually set the airspeed and Mach number indexes (figure 6-31). The landing airspeed index can be set over a range of 80 to 145 knots on the airspeed fixed dial. To set the index, rotate the knob clockwise or counterclockwise as necessary. The Mach limit index, which is adjustable from Mach number 0.5 to 1.0, should be set on the Mach rotating dial to the existing Mach limit. To set this index, push the knob in toward the case before rotating.

6-116. CHECKING AIRSPEED INDEX SETTING. To check the airspeed index (on the right-hand side of the case), rotate the MACH PUSH LIMIT knob without pushing it in. The airspeed index should move freely from 80 through 145 knots without sticking or binding. The Mach number index on the left-hand side of the dial should not move.

6-117. CHECKING MACH NUMBER INDEX SETTING. To check the Mach number index, push in and rotate the MACH PUSH LIMIT knob. The Mach number index should move from Mach number 0.5 through 1.0 without sticking or binding and the airspeed index should not move.

Figure No. 6-31. Airspeed and Mach Number Indexes

6-118. AIRSPEED AND MACH NUMBER INDICATOR DIAPHRAGM LEAK TEST. To check for diaphragm leakage, proceed as follows:

a. Remove airspeed and Mach number indicator from instrument panel. (Refer to paragraph 6-7.)

b. Disconnect pitot line from indicator and cover pitot line opening to keep out foreign matter.

c. Connect a source of pressure to the indicator and slowly apply sufficient pressure to cause approximately full scale deflection.

CAUTION

Never exceed full scale deflection of airspeed scale.

d. Close off pressure for a period of 30 seconds. Reading should not change more than 2 knots after the 30-second period.

e. Disconnect and remove source of pressure.

f. Attach pitot line to indicator.

g. Install indicator. (Refer to paragraph 6-8.)

6-119. AIRSPEED AND MACH NUMBER INDICATOR CASE LEAK TEST. To test for case leakage, proceed as follows:

a. Remove airspeed and Mach number indicator from instrument panel. (Refer to paragraph 6-7.)

b. Disconnect pitot and static lines from rear of indicator. Cover line openings to keep out foreign matter.

c. Join pitot and static pressure connections with a "Y" connection which, in turn, is connected to a mercury manometer and a source of suction.

d. Slowly apply a vacuum of 15 in. Hg and close off suction.

e. The manometer reading should not fall more than 0.4 in. Hg over a period of 10 seconds.

f. Disconnect source of suction, mercury manometer and "Y" connection from indicator.

g. Attach pitot and static pressure lines to indicator.

h. Install indicator. (Refer to paragraph 6-8.)

6-120. INDICATED AIRSPEED DIAL TEST AND INDICATOR POSITION ERROR TEST. To test indicated airspeed dial and to test position error, proceed as follows:

a. Remove airspeed and Mach number indicator from instrument panel. (Refer to paragraph 6-7.)

b. Disconnect pitot and static lines from rear of indicator. Cover line openings to keep out foreign matter.

c. Connect a source of pressure to pitot fitting on indicator. Leave static port open. Apply pressures according to table A; airspeed pointer should indicate speed specified within the tolerances given.

d. Disconnect and remove source of pressure.

e. Connect pitot and static lines to indicator.

f. Install indicator. (Refer to paragraph 6-8.)

Note

At the second test point on table A (21.538), take several readings with the instrument in different positions. Tap dial before each reading. The indication on the airspeed dial should not change more than 5 knots throughout the various positions.

TABLE A

PRESSURE (IN. WATER)	PRESSURE (IN. HG)	SPEED (KNOTS)	TOLERANCES SCALE	FRICTION (KNOTS)
5.310	0.390	90	± 3.0	5.0
21.538	1.583	180	± 3.5	5.0
61.820	4.543	300	± 5.0	5.0
187.843	13.803	500	± 7.0	5.0
287.404	21.119	600	± 8.0	5.0

6-121. MACH NUMBER DIAL TEST. To test the Mach number dial, proceed as follows:

a. Remove the airspeed and Mach number indicator from the instrument panel. (Refer to paragraph 6-7.)

b. Disconnect pitot and static lines from indicator. Cover line openings to keep out foreign matter.

c. Connect a source of suction to static port of indicator. Pitot port may be left open or pressure may be applied so that pointer can be used to aid in determining degree of alignment between the two dials.

d. Apply pressure altitude (suction) according to table B; the Mach number 1.0 mark on the Mach dial should align with the airspeeds specified within the tolerances given.

CAUTION

Never apply pressure to static pressure lines with instruments connected.

TABLE B

PRESSURE ALTITUDE (FEET)	INDICATED AIRSPEED AT MACH NUMBER 1.0 (IN KNOTS)	TOLERANCES SCALE	FRICTION (KNOTS)
5,000	613	± 12	5
10,000	566	± 11	5
30,000	390	± 8	5
50,000	249	± 5	5

e. Disconnect and remove source of suction from static port on indicator.

f. Connect pitot and static pressure lines to the indicator.

g. Install indicator. (Refer to paragraph 6-8.)

6-122. AIRSPEED CORRECTION CARD.

6-123. The airspeed correction card, which has a range from 80 through 650 knots, is provided as an aid to navigation. The card is mounted in a card holder which is attached to the canopy to the right of the instrument panel.

NAVIGATION INSTRUMENTS

6-124. NAVIGATION INSTRUMENTS.

6-125. The radio magnetic course indicator, ID-250A/ARN, and the course indicator, ID-249B/ARN, located on the pilot's instrument panel, present flight and navigational information to the pilot. Directional information is received at the indicators from three sources: (1) gyro-stabilized magnetic headings are obtained from the polar path compass system, (2) course information is received from either radio equipment, AN/ARN-14E (or radio equipment, AN/ARN-21), and (3) bearing information is received from radio equipment, AN/ARA-25. The indicators, polar path compass system and radio navigation equipment (figures 6-32 and 6-33) are electrically integrated in order to combine the bearing information from the radio receiving equipment with accurate magnetic headings for the purpose of aiding navigation. Heading-sensitive bearings can be obtained and a steady course can be maintained when the pilot utilizes the information presented by the two indicators. The bearing converter indicator, ID-251/ARN, in the radio system, AN/ARN-14E (or the azimuth indicator, ID-307/ARN, in the radio system, AN/ARN-21), located in the right-hand radio bay, is not used for visual reference but compares the bearing of the omni-range signal or TACAN signal (depending on which equipment is installed) to the magnetic heading to provide a course indication. The stand-by compass, located on top of the instrument panel shroud, indicates magnetic headings for emergency use only.

6-126. COURSE INDICATORS.

6-127. The radio magnetic course indicator, ID-250A/ARN, contains three synchronous motors, housed within the instrument case. The heading synchro is wired directly to the data transmitting autosyn in the compass coupler of the polar path compass system. The rotor of this synchro is linked mechanically to a rotating card on the face of the indicator which is calibrated to 360 degrees in 2-degree increments. As the card rotates, a fixed index at the top of the dial indicates the magnetic heading determined in the polar path compass system. This heading is independent of any radio equipment. Two indicating pointers on the face of the dial are marked "1" and "2" and are linked to the rotors of the other two synchros. The synchro which is linked to pointer No. 2 is wired to the differential synchro in the bearing converter indicator, ID-251/ARN (or the azimuth indicator, ID-307/ARN, if radio equipment, AN/ARN-21, is installed), to indicate the magnetic bearing to the station tuned in on radio equipment, AN/ARN-14E (or AN/ARN-21), from the airplane's position. Pointer No. 1 is linked to the rotor of the third synchro within the case and indicates the magnetic bearing to the station from the airplane as determined by the radio equipment, AN/ARA-25. When "ADF" is not selected on the uhf control panel, pointer No. 1 will be slaved to pointer No. 2. The course indicator, ID-249B/ARN, consists of a cross-pointer indication, a relative heading indicating pointer, an observation window (FROM-TO) and a course window, all of which are visible on the face of the dial. On the front of the case in the lower left-hand corner is a course selector knob (SET) and in the upper right-hand corner is a marker beacon light. Two flag alarms (OFF), one on the vertical line and one on the horizontal line, are provided for the cross-pointer indication. The flag will appear when the line is not receiving information from the radio receiver. Housed within the instrument case are a course selector resolver and a heading synchro which are wired in parallel with the data transmitting autosyn of the compass coupler in the polar path compass system. The vertical pointer of the cross-pointer indication indicates lateral deviation from a selected omni-range or TACAN course as determined by the omni-range or TACAN radio navigation equipment (AN/ARN-14E or AN/ARN-21) and set in the course window. The information is presented on the No. 2 pointer of the radio magnetic course indicator, ID-250A/ARN. The pilot then manually sets the course in the course window of the course indicator, ID-249B/ARN, by operating the course selector knob. After the course is thus set in the course window, the vertical line indication will be centered when the airplane is on course. Whenever the vertical line is off-center, indicating the airplane is off course, the airplane heading is changed in the direction of the line to resume course. The relative heading pointer, which is connected to the heading synchro, indicates the angle between the magnetic heading and the course set into the course window. The pointer moves 360 degrees around the inside edge of the circular face of the instrument and its travel is calibrated to 45 degrees each side of center and also top and bottom. In the absence of wind, the pointer and the vertical line will be centered at the zero position, indicating on course. When a cross wind exists, the mechanical procedure of keeping the heading pointer under the centered vertical line will not give proper drift correction. A cross wind exists if keeping the heading pointer under the vertical line does not maintain the desired track or when the vertical line does not continue to show a rate of movement toward the center while intercepting a bearing. When this occurs, the pilot must compute and make a large correction to intercept the desired bearing. As the vertical line centers, the path of the airplane is toward the desired bearing, corrected for drift. The drift correction will be shown on the indicator by the deflection of the heading pointer from the center of the instrument. The FROM-TO window, located at the upper left of the instrument, indicates the geographical location of the airplane with respect to a line passing through the range station, perpendicular to the course set into the course window.

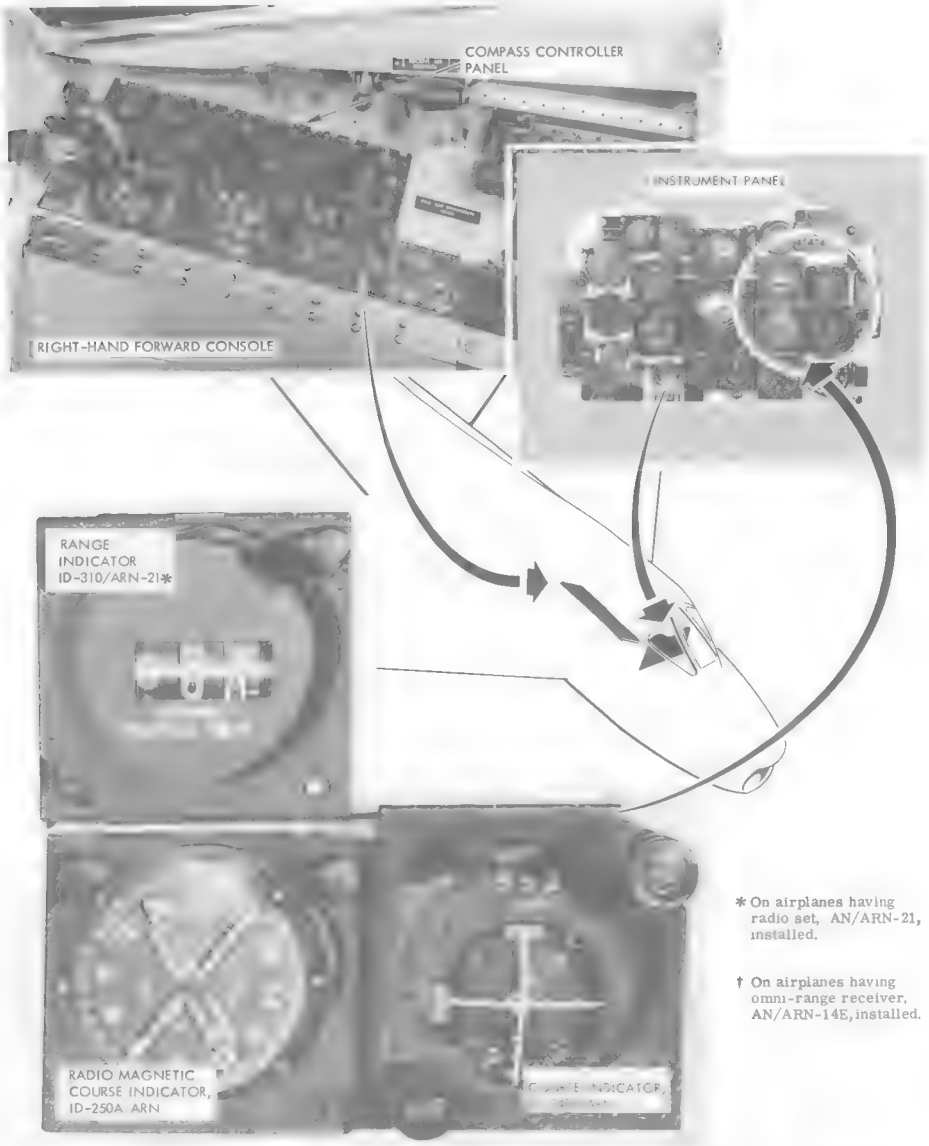

Figure No. 6-32. Course Indicators and Polar Path Comoass System (Sheet 1)

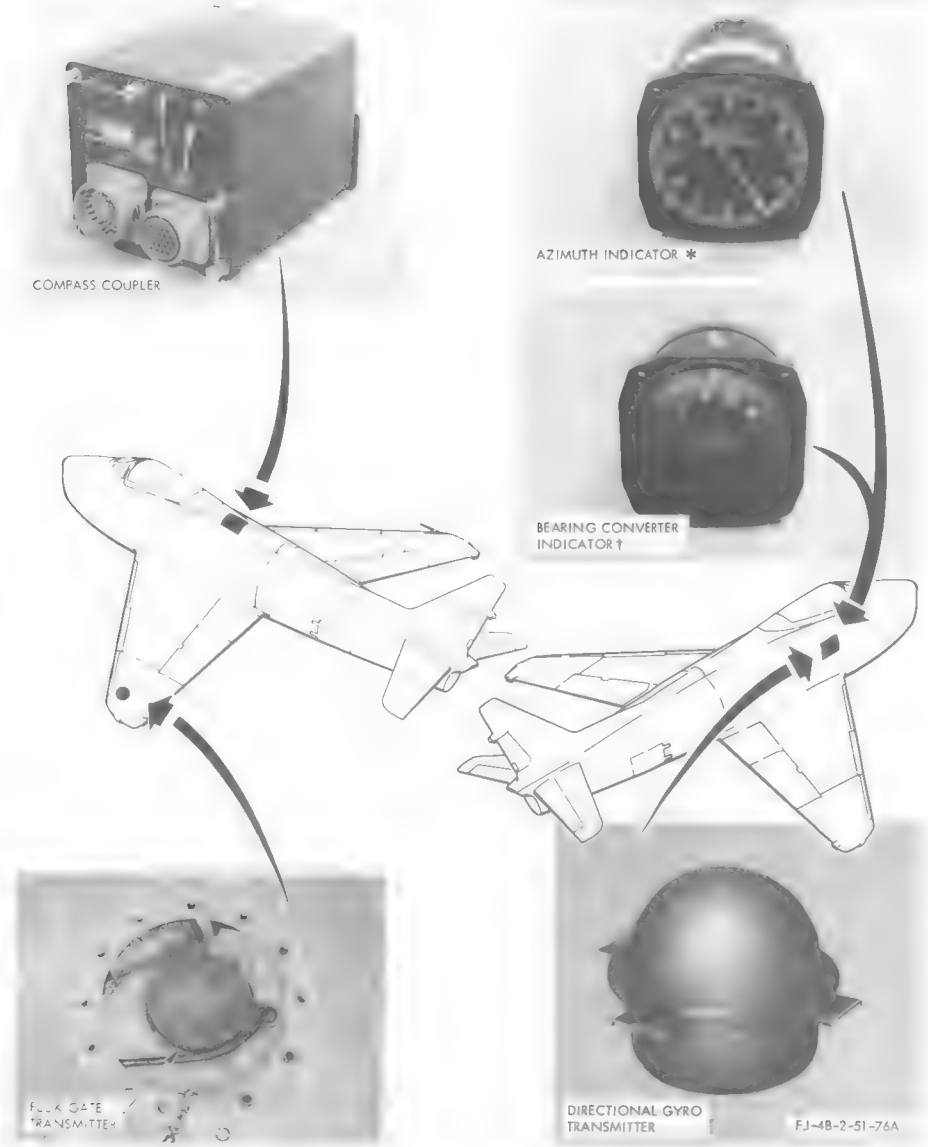

Figure No. 6-32. Course Indicators and Polar Path Compass System (Sheet 2)

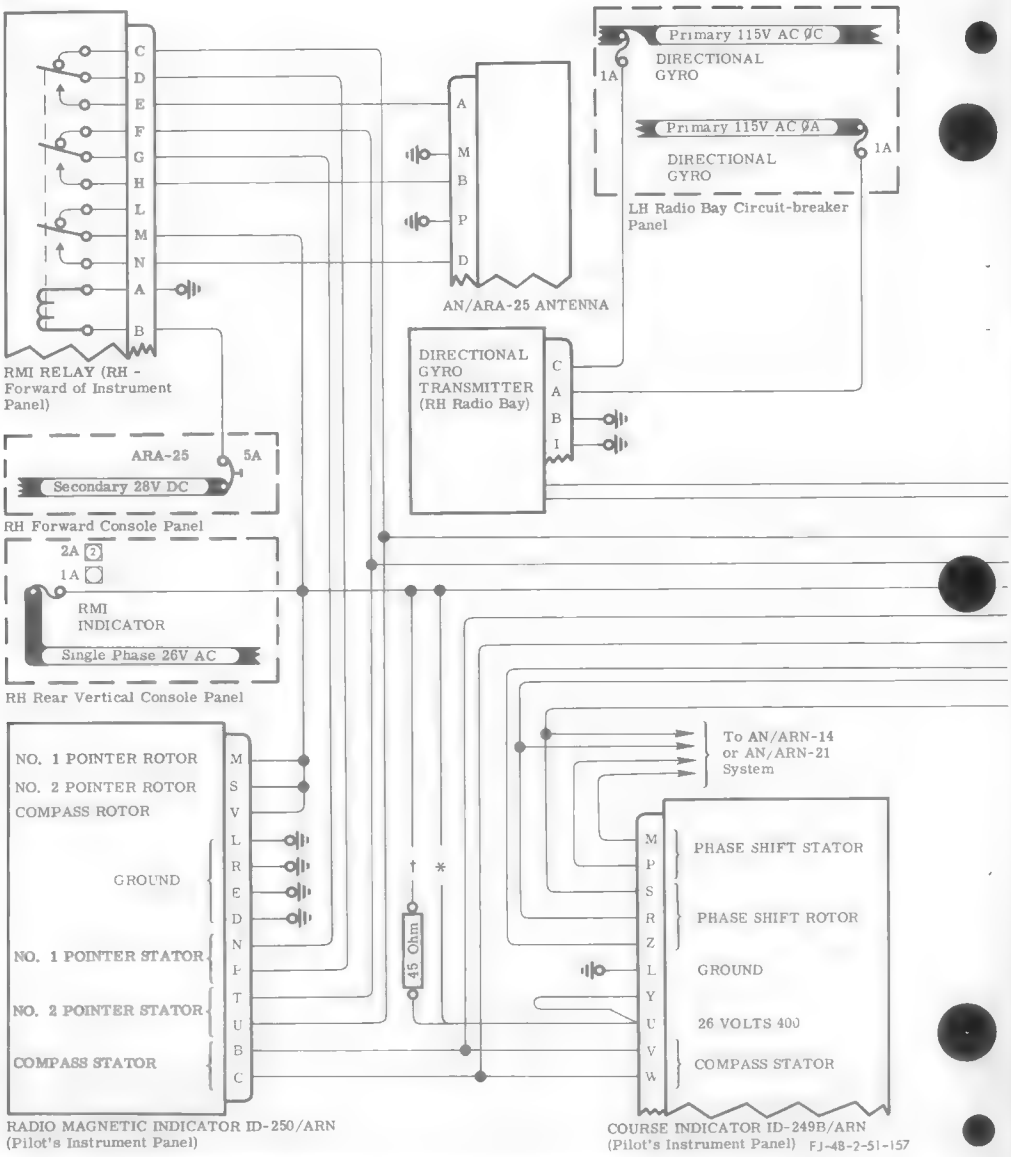

Figure No. 6-33. Course Indicators and Polar Path Compass System Schematic (Sheet 1)

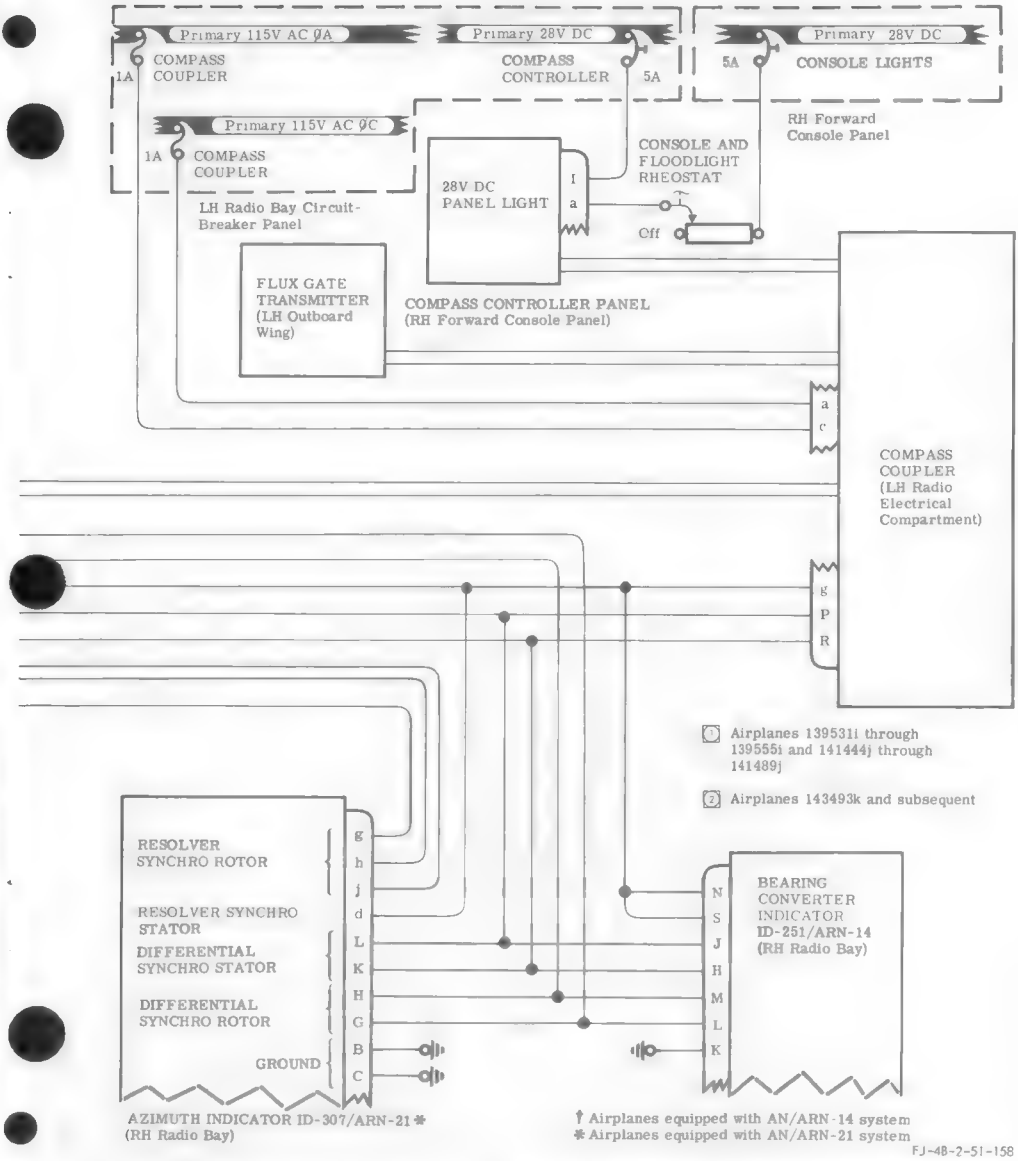

Figure No. 6-33. Course Indicators and Polar Path Compass System Schematic (Sheet 2)

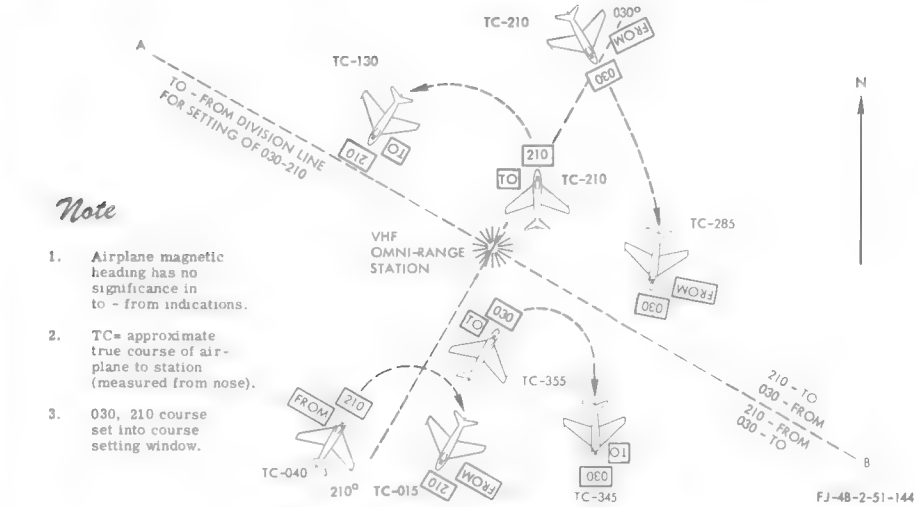

Figure No. 6-34. Course Indicator Ambiguity Indications

Specifically, the indication will be "TO" if the course set into the course window is within ±90 degrees of the airplane's true course to the station; the indication will be "FROM" if the course set into the course window is more than ±90 degrees from the airplane's true course to the station. (See figure 6-34.) The horizontal line of the cross-pointer indication is for use with a glide path receiver to indicate the final approach angle, but this receiver is not used in this airplane. The bearing converter indicator, ID-251/ARN, or the azimuth indicator, ID-307/ARN (depending upon which radio system is installed), contains a rotating azimuth scale, a heading synchro, wired to the data transmitting autosyn in the compass coupler, and a differential synchro. Bearing information is visually displayed on pointer No. 2 at the radio magnetic course indicator, ID-250A/ARN. The range indicator, ID-310/ARN, which is used only with the radio navigation equipment, AN/ARN-21, indicates the distance from the airplane to the surface beacon. A single window on the indicating dial indicates this information in nautical miles. While the indicator is searching for the correct range, the numbers in the indicating window will rotate very rapidly. To guard against the pilot reading incorrect distance indications, the indicating dial is partially covered with a red flag until the range is determined. Power for the course indicators is from the 26-volt, single-phase a-c bus. On airplanes 139531i through 141489j, the circuit is protected by a one-ampere fuse (RMI IND) located on the right-hand rear vertical console panel. On airplanes 143493k and subsequent, the circuit is protected by a two-ampere fuse (RMI IND) located on the right-hand rear vertical console.

6-128. POLAR PATH COMPASS SYSTEM.

6-129. The polar path compass system (figures 6-32 and 6-33) furnishes a continuous accurate directional reference under any flight condition and regardless of latitude. The system consists of a miniature flux gate transmitter, a low-random drift directional gyro transmitter, a compass coupler and a console controller panel. Three modes of operation, selected by means of a selector switch on the controller panel, are possible to provide the required type of directional reference: (1) the directional gyro alone, (2) the directional gyro and the flux gate transmitter and (3) the flux gate transmitter alone. During flight in high latitude regions, rapidly converging meridians necessitate constant changes in magnetic references. Also, the compass headings established by magnetic sensing are unreliable. In these areas, the low-drift directional gyro transmitter alone is employed to provide reference headings for the airplane to fly a great circle course. In regions where magnetic sensing is reliable, the second mode of operation is selected. The output of the directional gyro transmitter is varied by the magnetic heading information determined in the flux gate transmitter. This mode combines the advantages of both types of systems to provide a stable, accurate and reliable heading. The third mode of operation cuts out the directional gyro and associated circuitry and provides a source of stand-by compass information from the flux gate transmitter. Visual indications of reference headings determined by any mode

of operation in the polar path compass system are read on the rotating dial of the radio magnetic course indicator, ID-250A/ARN, located on the pilot's instrument panel. Directional reference information from the polar path compass system is also fed to the radio and omnidirectional indicators where it combines with bearing information to determine course. Alternating current for the compass system is from the 115-volt, 400-cycle, phase "A" and phase "C" primary busses. These circuits are protected by two one-ampere fuses (COMPASS COUPLER and DIRECTIONAL GYRO) for each phase. Direct current for the compass system is from the primary 28-volt d-c bus; this circuit is protected by a 5-ampere circuit breaker (COMPASS CONTROLLER). The four fuses and the circuit breaker are located on the left-hand radio bay circuit-breaker panel.

6-130. FUNCTION OF POLAR PATH COMPASS SYSTEM — DIRECTIONAL GYRO MODE OF OPERATION.

6-131. The gyro of the directional gyro transmitter, which is never directionally slaved or mechanically positioned, provides a stabilized directional reference for the compass system. When the system is energized and "D.G." is selected by the selector switch located on the compass controller panel (figures 6-32 and 6-35), the gyro spin axis assumes an arbitrary position which is used as a fixed reference. The heading indication on the rotating card of the radio magnetic course indicator has no geographical significance until the pilot sets the course by operating the course setting knob (COURSE SET) located on the compass controller panel. The stator of the directional gyro transmitting autosyn, located in the directional gyro transmitter, is wired back to back with the stator of the directional gyro follow-up autosyn located in the compass coupler. The rotor of this follow-up autosyn is mechanically coupled to the output data shaft, and its electrical output feeds the gyro amplifier. The motor, in turn, rotates the output data shaft through a gear train, positioning the rotor of the data transmitting autosyn. (The stator of the data transmitting autosyn is wired back to back with the stator of the heading synchro in the indicator.) Thus, changes in heading will produce proportional changes in the position of the output data shaft, resulting in rotation of the compass card at the indicator. A separate servo system is provided for the purpose of setting the course on the indicator and for making gyro apparent drift corrections. The stator of the directional gyro follow-up autosyn is mechanically coupled, through a large gear reduction, to the slaving or correcting servomotor. When the course setting knob is pushed and turned, the course potentiometer feeds a proportional electrical signal into this slaving servo system. The stator of the directional gyro follow-up autosyn is rotated at a speed proportional to knob rotation, the output shaft is rotated and, subsequently, the desired heading is indicated on the compass card. When the latitude knob, which provides a method of correcting the drift rate, is turned, the stator of the directional gyro follow-up autosyn is rotated by the servomotor. The motor is controlled by the signal from the drift rate potentiometer in the compass controller panel which is routed through the amplifier in the compass coupler. The speed of the output data shaft is proportional to the drift rate potentiometer signal.

6-132. FUNCTION OF POLAR PATH COMPASS SYSTEM — DIRECTIONAL GYRO AND FLUX GATE TRANSMITTER.

6-133. When the slaved directional gyro mode of operation (SLAVED) is selected on the compass controller panel, the directional gyro and the mechanism which servoes the output data shaft in proportion to directional gyro heading displacement still form the basic function of the system. When this mode of operation is first selected, the signal from the directional gyro provides a short-term stabilized reference on the indicator. However, with the flux gate tied into the system, the output data shaft is corrected by flux gate signals through the stator of the directional gyro follow-up autosyn, regardless of the directional gyro heading signals. The stator of the flux gate transmitter autosyn is wired back to back with the flux gate follow-up autosyn in the compass coupler. (See figure 6-35.) The rotor of the flux gate follow-up autosyn is mechanically coupled to the output data shaft through the cam-type deviation compensator. The electrical output of the rotor feeds the same amplifier that is used for gyro drift rate correction in the "D.G." mode of operation. Therefore, the same motor generator and large gear reduction move the stator of the directional gyro follow-up autosyn in accordance with magnetic signals. The stator will move slowly over a long period of time to actuate the servo system and, at the same time, the output data shaft will mechanically drive the flux gate follow-up autosyn until a null is reached. Normal or "slow slaving" action occurs at the rate of approximately ½ degree per minute. When the output data shaft and indicator heading are synchronized with the magnetic heading of the flux gate transmitter, the corrective action ceases. The output data shaft may be synchronized with the magnetic heading very rapidly by means of the automatic synchronization knob (PUSH-SYNC.) located on the compass controller panel. Pushing the knob in will operate a magnetic clutch in the compass coupler that will shift gears in the slaving servo system. The slaving, or correcting to synchronization, will occur at a rate of from maximum misalignment to synchronization in less than 20 seconds. At any time the slaved "D.G." mode of operation is selected, the degree to which the output data shaft is synchronized to magnetic heading will be indicated on the annunciator, or synchronizer indicator, located just above the PUSH-SYNC. knob on the compass controller panel. When synchronization is achieved, the indicating needle will be centered. The COURSE SET knob, when it is depressed and turned, may be used to synchronize the system. The same operation takes place when utilizing this knob as when using the PUSH-SYNC. knob.

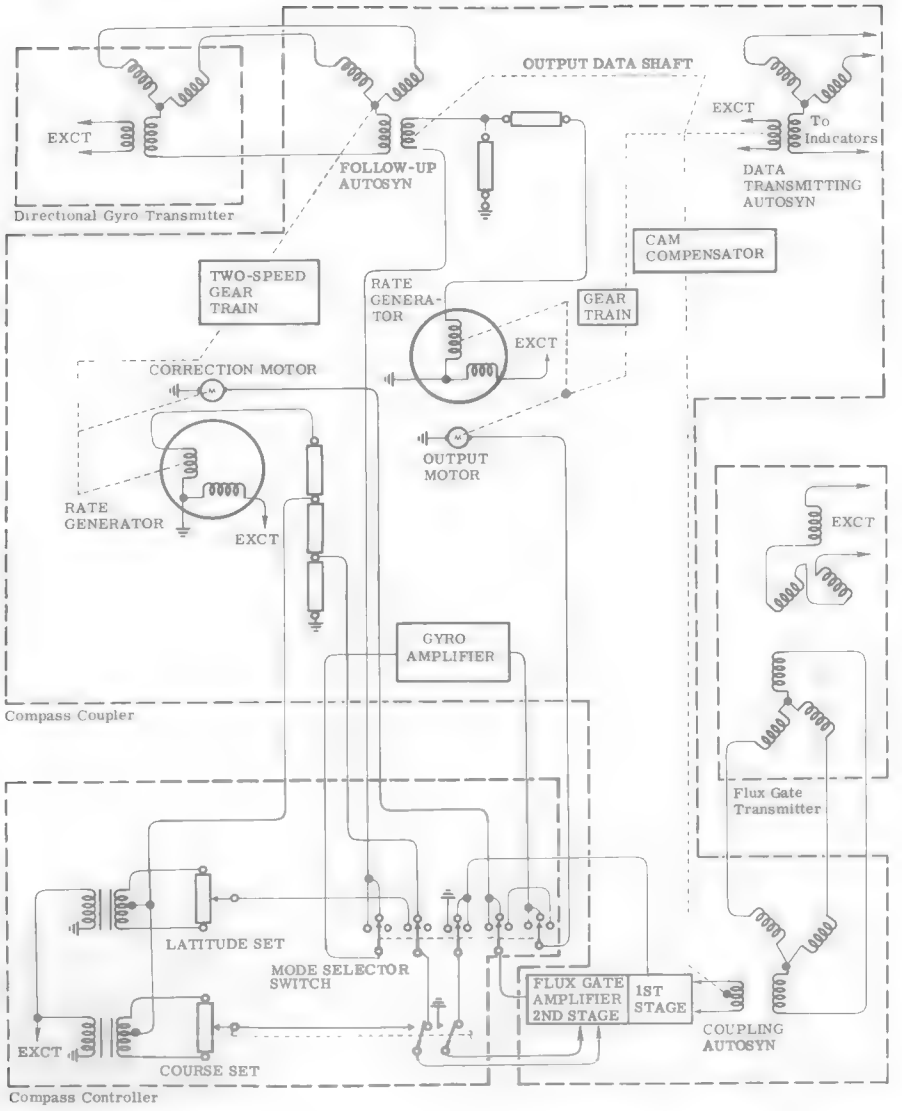

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Figure No. 6-35. Polar Path Compass System Functional Schematic

6-134. FUNCTION OF POLAR PATH COMPASS SYSTEM — FLUX GATE MODE OF OPERATION.

6-135. When the selector switch on the compass controller panel is placed in the "COMP" position, only the equipment essential to a basic flux gate compass system is utilized. (See figure 6-35.) Signals from the rotor of the flux gate follow-up autosyn feed the flux gate amplifier in the compass coupler. The output of the flux gate amplifier then feeds into the motor generator unit normally actuated by the directional gyro. The output data shaft is then displaced proportionately and rapidly by flux gate heading signals.

6-136. TROUBLE SHOOTING POLAR PATH COMPASS SYSTEM.

6-137. Before performing any trouble shooting procedures on the compass system, or replacing any component of the system, it should first be determined if the source of trouble is in the wiring or power supply. A continuity check of all wiring should be accomplished. The power supply voltage and phase rotation should be checked at the directional gyro transmitter and the compass coupler. (Refer to paragraph 6-138.)

CAUTION

Do not at any time attempt to open or disassemble the directional gyro transmitter or the flux gate transmitter.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
HEADING INFORMATION IS INACCURATE ON "D.G." MODE BUT IS ACCURATE ON OTHER MODES OF OPERATION.		
Directional gyro has excessive random drift.		Replace directional gyro transmitter. (Refer to paragraph 6-147.)
Latitude potentiometer selector switch or transformer of the compass controller defective.		Replace the compass controller panel. (Refer to paragraph 6-153.)
Rate generator of coupler correction motor-generator set defective.		Replace the compass coupler. (Refer to paragraph 6-150.)
HEADING INFORMATION INACCURATE ON BOTH "D.G." AND "SLAVED" MODES BUT IS ACCURATE ON "COMP" MODE OF OPERATION.		
Defective directional gyro amplifier in compass coupler.		Replace the compass coupler. (Refer to paragraph 6-150.)
Defective directional gyro transmitter.		Replace the directional gyro transmitter. (Refer to paragraph 6-147.)
Defective directional gyro follow-up autosyn in the compass coupler.		Replace the compass coupler. (Refer to paragraph 6-150.)
Excessive friction in correction motor gear train or a defective correction motor.		Replace the compass coupler. (Refer to paragraph 6-150.)
HEADING INFORMATION ACCURATE ON "D.G." MODE, BUT SYSTEM SLAVES TO INACCURATE HEADINGS ON "SLAVED" MODE AND IS INACCURATE ON "COMP" MODE OF OPERATION.		
A spurious magnetic field has been brought near the flux gate transmitter.	Remove the source of trouble.	If trouble is not corrected, the flux gate has been damaged. Replace the flux gate transmitter. (Refer to paragraph 6-144.)
Deviation compensation has been incorrectly performed.	Perform compass swinging and compensation procedures. (Refer to paragraph 6-35.)	If trouble is not corrected, replace the flux gate transmitter. (Refer to paragraph 6-144.)

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
HEADING INFORMATION ACCURATE ON "D.G." MODE, BUT SYSTEM SLAVES TO INACCURATE HEADINGS ON "SLAVED" MODE AND IS INACCURATE ON "COMP" MODE OF OPERATION. (Cont)		
Defective flux gate transmitter.		Replace the flux gate transmitter. (Refer to paragraph 6-144.)
Defective flux gate follow-up autosyn in the compass coupler.		Replace the compass coupler. (Refer to paragraph 6-150.)
HEADING INFORMATION INACCURATE ON ALL MODES OF OPERATION.		
Defective flux gate amplifier in the compass coupler.		Replace the compass coupler. (Refer to paragraph 6-150.)
Defective data transmitting autosyn in the compass coupler.		Replace the compass coupler. (Refer to paragraph 6-150.)
Excessive friction in the output motor gear train.		Replace the compass coupler. (Refer to paragraph 6-150.)
HEADING INFORMATION OSCILLATES.		
Defective electrical connections in output motor-generator set in the compass coupler.		Replace the compass coupler. (Refer to paragraph 6-150.)
Defective radio magnetic course indicator, ID-250A/ARN.		Replace indicator. (Refer to paragraphs 6-7 and 6-8.)
Defective electrical wiring.	Check shield and ground leads for open circuits.	Correct open circuit.
LOSS OF HEADING INFORMATION ON "D.G." OR "SLAVED" MODE BUT NOT ON "COMP" MODE OF OPERATION.		
Defective tube in directional gyro amplifier in the compass coupler.		Replace the compass coupler. (Refer to paragraph 6-150.)
Open rotor lead in directional gyro follow-up autosyn or open lead in rate generator of output servomotor-generator set, both located in the compass coupler.		Replace the compass coupler. (Refer to paragraph 6-150.)
Open circuit in rotor of the directional gyro autosyn.		Replace directional gyro transmitter. (Refer to paragraph 6-147.)
Defective directional gyro amplifier in the compass coupler.		Replace the compass coupler. (Refer to paragraph 6-150.)
Defective selector switch on the compass controller panel.		Replace the compass controller panel. (Refer to paragraph 6-153.)

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
LOSS OF HEADING INFORMATION ON ALL MODES OF OPERATION.		
Open circuit rotor of the data transmitter autosyn or open circuit in output motor winding of the compass coupler.		Replace the compass coupler. (Refer to paragraph 6-150.)
Defective selector switch on the compass controller panel.		Replace the compass controller panel. (Refer to paragraph 6-153.)
EXTREMELY SLUGGISH OPERATION ON "D.G." AND "SLAVED" MODES OF OPERATION.		
Open circuit in stator of directional gyro follow-up autosyn in compass coupler.		Replace the compass coupler. (Refer to paragraph 6-150.)
Open circuit in stator of directional gyro autosyn.		Replace the directional gyro transmitter. (Refer to paragraph 6-147.)
SYNCHRONIZING INDICATOR SHOWS NO MOVEMENT.		
Defective flux gate amplifier in the compass coupler.		Replace the compass coupler. (Refer to paragraph 6-150.)
Loss of excitation to the compass controller panel.	Check connection between pin "C" on the compass coupler to pin "S" on the compass controller panel. Check a-c power supply to the compass coupler. (Refer to paragraph 6-138. Check operation of synchonizer indicator using test compass coupler in system.	Correct open circuit. If trouble is corrected, replace compass coupler. (Refer to paragraph 6-150.)
Defective synchronizer indicator circuit.		Replace the compass controller panel. (Refer to paragraph 6-153.)
INABILITY TO SYNCHRONIZE IN "SLAVED" MODE.		
Loss of d-c excitation through synchronizing and course set circuits in controller to the magnetic clutch in the compass coupler.	Check connection between pin "f" on the compass controller and pin "Z" on the compass coupler. Check d-c power supply to pin "i" on the compass controller panel. Check 5-ampere circuit breaker (COMPASS CONTROLLER) located on the left-hand radio bay circuit-breaker panel.	Correct open circuit. Make connections as necessary. Close circuit breaker. If trouble persists, replace the compass controller panel. (Refer to paragraph 6-153.)
Defective magnetic clutch in the compass coupler.		Replace the compass coupler. (Refer to paragraph 6-150.)
Defective synchronizer switch in the compass controller.		Replace the compass controller panel. (Refer to paragraph 6-153.)
Defective tube in the flux gate amplifier in the compass coupler.	Switch from "SLAVED" mode to "COMP" mode of operation by means of the selector switch on the compass controller panel.	If the system operates correctly, this is not the source of trouble. If the system does not operate correctly, replace the compass coupler. (Refer to paragraph 6-150.)

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
INABILITY TO SYNCHRONIZE IN "SLAVED" MODE. (Cont)		
Defective flux gate amplifier in the compass coupler.	Switch from "SLAVED" mode to "COMP" mode of operation by means of the selector switch.	If the system operates correctly, this is not the source of trouble. If the system does not operate correctly, replace the compass coupler. (Refer to paragraph 6-150.)
Open circuit in the correction or slaving motor in the compass coupler.		Replace the compass coupler. (Refer to paragraph 6-150.)
Open circuit in the rotor of the flux gate follow-up autosyn in the compass coupler.	Switch from "SLAVED" mode to "COMP" mode of operation by means of the selector switch.	If the system operates correctly, this is not the source of trouble. If the system does not operate correctly, replace the compass coupler. (Refer to paragraph 6-150.)
Defective flux gate transmitter.	Switch from "SLAVED" mode to "COMP" mode of operation by means of the selector switch.	If the system operates correctly, this is not the source of trouble. If the system does not operate correctly, replace the flux gate transmitter. (Refer to paragraph 6-144.)
SLUGGISH "COURSE SET" OPERATION.		
Refer to first six probable causes listed under trouble INABILITY TO SYNCHRONIZE IN "SLAVED" MODE.		
Loss of a-c excitation to the compass controller.	Check connection from pin "C" on the compass coupler to pin "S" on the compass controller and check connection from pin "D" on the compass coupler to pin "T" on the compass controller.	If connections are good, replace the compass coupler. (Refer to paragraph 6-150.)
Defective "COURSE SET" potentiometer in the compass controller.		Replace the compass controller panel. (Refer to paragraph 6-153.)

6-138. CHECKING POLAR PATH COMPASS SYSTEM ALTERNATING-CURRENT POWER SUPPLY.

6-139. For the correct operation of the polar path compass system, 115 (±5) volt, 400 (±20) cycle, three-phase alternating current must be supplied to the system. The correct phase rotation must be "ABC" with "B" grounded. Voltage and phase rotation should be checked periodically. To check voltage, proceed as follows:

a. Unscrew the four one-ampere fuses (two COMPASS COUPLER fuses and two DIRECTIONAL GYRO fuses), located on the left-hand radio bay circuit-breaker panel.

b. Remove caps and insert fuses.

c. Place the d-c power switch, located on the right-hand forward console, in the "OFF" position.

d. Connect external power to airplane.

e. Position the INST. AC POWER switch to the "NO. 1 INV." position.

f. Using an a-c voltmeter, check for 115 volts at each fuse between the exposed end of the fuse and ground. The phase rotation should be checked at the cable plugs of the compass coupler and the directional gyro transmitter after the cabling has been connected to the power source. Connect a phase sequence tester to the cable plugs; the correct phase sequence is indicated if the lamp illuminates. (See figure 6-36.)

6-140. OPERATIONAL CHECK OF POLAR PATH COMPASS SYSTEM.

6-141. To perform an operational check of the polar path compass system, proceed as follows:

a. Start the engine and run at idle rpm. (Refer to paragraph 1-9.)

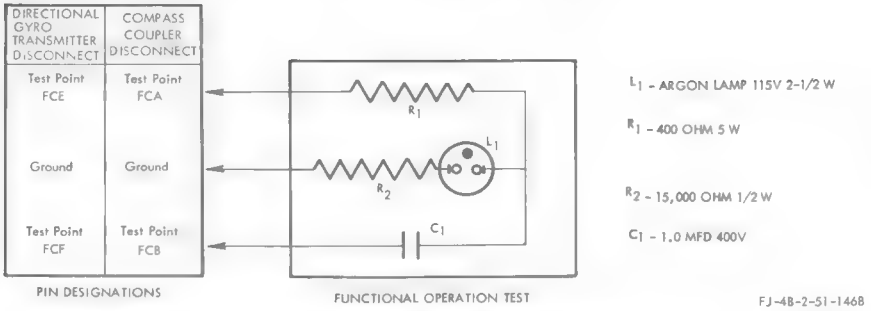


Figure No. 6-36. Polar Path Compass System Phase Sequence Tester Circuit

b. Allow at least 3 minutes for the compass system to warm up.

c. Set the latitude correction dial on the compass controller panel to the local latitude.

d. Set the system to "SLAVED" by means of the selector switch.

e. Push the PUSH-SYNC. switch until the meter, located above the switch on the panel, indicates a null.

f. The uncompensated and the compensated dials on the compass coupler and the compass card of the radio magnetic course indicator should agree and indicate the approximate magnetic heading of the airplane.

g. Switch the selector switch to "COMP" mode of operation.

h. Observe the radio magnetic course indicator. There should be no change in compass heading on the compass card of the indicator.

i. Return the system to "SLAVED" by means of the selector switch.

j. Operate the COURSE SET knob to increase the compass heading indication on the radio magnetic course indicator. The readings on the dials on the compass coupler should also increase.

k. Return the system to null by means of the COURSE SET knob.

Note

The COURSE SET knob must be turned in the same direction that the annunciator needle must move to return to a null indication.

l. Change the airplane heading approximately 180 degrees in azimuth.

m. The compass indication on the card of the radio magnetic course indicator should follow this change in heading.

n. The time required for the system to return to a null by operation of the COURSE SET knob or the PUSH-SYNC. knob should be less than 20 seconds.

o. Push in and turn COURSE SET knob to the right to increase the reading of the radio magnetic course indicator by approximately 5 degrees from the null. Release the knob. The synchronizing indicator, or annunciator, should be deflected to the right. The radio magnetic course indicator should return to the original indication (null) at a rate of about ½ degree per minute or 5 degrees in 10 minutes. The annunciator should have returned to the centered position at the end of the 10-minute period.

p. Select the "D.G." mode of operation by means of the selector switch on the panel.

q. Operate the COURSE SET knob to set a heading of "0" on the radio magnetic course indicator.

r. Set the latitude correction dial to 90 degrees south latitude.

s. Depress the PUSH-SYNC. knob on the compass controller panel.

t. The reading of the radio magnetic course indicator should decrease at a rate of about 350 degrees per minute.

u. With "D.G." still selected, change the LATITUDE dial to indicate the local latitude.

v. Allow time for the heading to stabilize and then accurately note the reading of the radio magnetic course indicator.

w. Allow the system to operate for 30 minutes without moving the airplane or disturbing any system components.

x. At the end of the 30-minute period, the radio magnetic course indicator reading should not have changed by more than ±2 degrees.

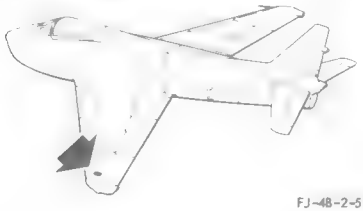
6-142. POLAR PATH COMPASS SYSTEM—FLUX GATE TRANSMITTER.

6-143. The flux gate transmitter generates output voltages which represent the magnetic heading of the airplane. This heading is used for the magnetic reference in both the "SLAVED" and the "COMP" mode of operation. The transmitter is mounted in the left wing near the tip, removed from disturbing magnetic influences. The basic components of the transmitter are a flux gate element, gimbaled in both pitch and bank, and a compass float, containing two magnets, pendulously mounted in pitch and bank and free in azimuth. The flux gate element, because of its gimbaled mounting, will remain horizontal to the earth when the airplane pitches and banks but will change direction with the airplane. The compass card, mounted on top of a gimbal, also remains horizontal to the earth when the airplane pitches and banks. The two magnets within the compass card align themselves with the horizontal component of the earth's magnetic field, strengthening the field and providing a more uniform field than is normally available. This results in a more reliable magnetic heading reference at lower latitudes than would normally be possible. The operation of the flux gate transmitter is based on the principle that magnetic lines of force flow more easily through a material of low reluctance than through air. Consequently, a coil wound on a core of metal with low reluctance and placed in the earth's magnetic field will allow the magnetic lines of force to be concentrated in this core. The flux gate element consists of three cores with excitation and output coils which are physically arranged to form an equilateral triangle with the output coils "Y" connected. In order to induce voltages in these coils, a-c current is passed through each coil in the winding. During periods of saturation, the magnetic lines of force are rejected and, during periods of unsaturation, the magnetic lines of force are admitted and cut the coils, inducing voltages. The induced voltages are dependent upon the position of the flux gate in azimuth. Thus, a different set of voltages is induced for each magnetic heading of the airplane.

6-144. REMOVING AND INSTALLING FLUX GATE TRANSMITTER.

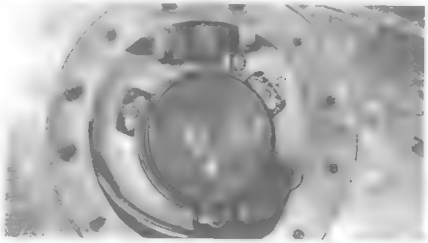
REMOVING

1 Remove circular compass access panel on top of left outboard wing.



FJ-48-2-51-77

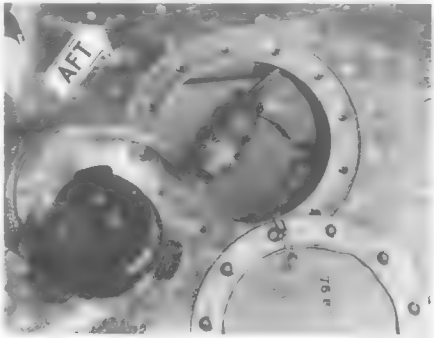
2 Note position of the graduated scale on mounting tab with respect to the fixed reference mark on the support assembly.



Note During swinging and compensating procedures, the flux gate transmitter is realigned when necessary, by loosening the mounting screws and rotating the unit to compensate for index error. To avoid excessive index errors after installing a new flux gate transmitter, the fixed reference mark should be used as a guide.

3 Remove three mounting screws that secure transmitter to support assembly. Lift transmitter from support assembly and lay aside on upper wing surface.

4 Remove clamp from connector, disconnect pigtail from airplane cable and remove transmitter from airplane.



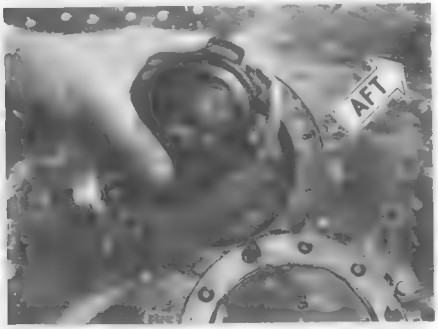
FJ-48-2-51-102

INSTALLING

1 Connect pigtail connector of flux gate transmitter to airplane electrical cable and safety-wire with AN995F32 lockwire.

2 Clamp electrical connector to support assembly.

3 Position transmitter in support assembly making sure the graduated scale is aft.



4 Install the three mounting screws. Align the zero on graduated scale with the fixed reference on the support assembly upper surface and then tighten mounting screws.

5 Perform an operational check of polar path compass system. (Refer to paragraph 6-140.)

6 If a new flux gate transmitter is installed, perform swinging and compensating procedures for the polar path compass system, steps A through G. (Refer to paragraph 6-169.)

7 Replace access panel.

FJ-48-2-51-190

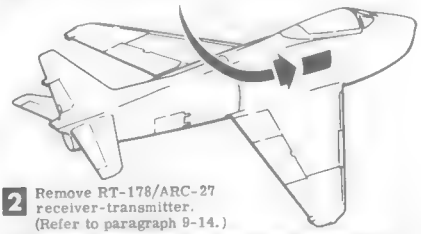
6-145. POLAR PATH COMPASS SYSTEM — DIRECTIONAL GYRO TRANSMITTER.

6-146. The directional gyro transmitter provides autosyn output signals which furnish an inertia reference and which represent the changes in the azimuth position of the airplane. The unit is shock-mounted on a support bracket, located in the right-hand radio bay, forward of the receiver-transmitter, RT-178/ARC-27. Within the directional gyro transmitter, a low-drift, horizontal free gyro stabilizes an autosyn transmitter which is sensitive to changes in azimuth of the gyro housing. The gyro is an induction motor which consists of a rotor that rotates about a stator and which is gimbaled and mounted on pivots so as to be completely free in all three axes. The mounting arrangement is such that the gyro and both gimbals will remain rigidly fixed in space while the gyro housing turns in azimuth. The autosyn rotor is attached to a rigid gimbal of the gyro while the autosyn stator is mounted on the gyro housing. Thus, as the airplane turns in azimuth and the gyro housing turns with it, the autosyn stator changes position relative to the autosyn rotor. An autosyn output voltage is thus developed which represents this change in azimuth. When it is originally energized, the gyro is erected to a horizontal position very rapidly by acceleration forces but, when this action ceases, an a-c, d-c erection mechanism maintains the spin axis of the gyro in a horizontal position to reduce random gyro drift.

6-147. REMOVING AND INSTALLING DIRECTIONAL GYRO TRANSMITTER.

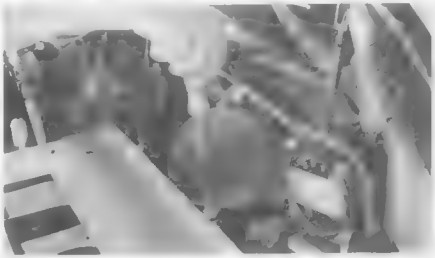
REMOVING

1 Open right-hand radio bay access door.



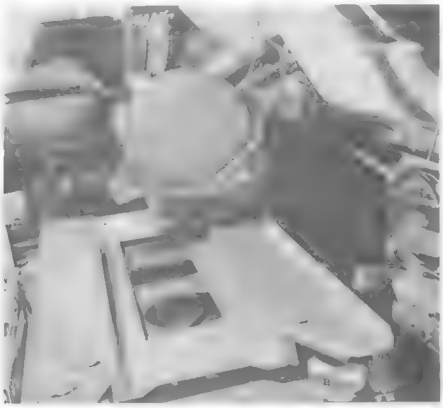
2 Remove RT-178/ARC-27 receiver-transmitter. (Refer to paragraph 9-14.)

3 Remove four screws and eight washers that secure the mounting tabs on the gyro to the individual shock mounts on the mounting bracket.



Note A bond jumper, located at the left forward shock mount, is removed from the mounting tab with the screw.

4 Lift gyro up and out of mounting and set gyro temporarily on RT-178/ARC-27 receiver-transmitter mount.



5 Cut safety wire and disconnect gyro pigtail connector from plug on airplane cable.

6 Gather cable and connector at mounting tab to avoid hanging up electrical connector when removing.

7 Hold gyro by two mounting tabs and lift gyro up and out of the airplane.

INSTALLING

1 Place gyro temporarily on the RT-178/ARC-27 receiver-transmitter mount.

2 Make electrical connection and safety-wire connector with AN995F32 lockwire.

3 Position gyro in mounting hole and align holes in mounting tabs with holes in shock mounts.

4 Install bond jumper on top of left-hand forward mounting tab.

5 Secure gyro with eight washers and four screws.

6 Install RT-178/ARC-27 receiver-transmitter. (Refer to paragraph 9-14.)

7 Close and secure access door.

8 Perform an operational check of polar path compass system. (Refer to paragraph 6-140.)

FJ-48-2-51-104

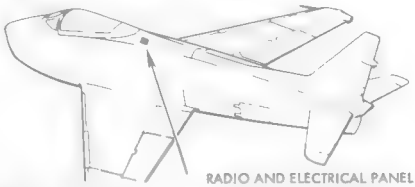
6-148. POLAR PATH COMPASS SYSTEM—COMPASS COUPLER.

6-149. The compass coupler houses all of the servomechanical mechanisms and servo amplifiers which are provided to convert the signals from the flux gate transmitter and the directional gyro transmitter into accurate directional information at all latitudes. The unit contains four major assemblies: (1) the power supply, (2) the flux gate servo amplifier, (3) the directional gyro servo amplifier and (4) the mechanism assembly. The power supply includes a three-phase power transformer, a B-power supply consisting of a three-phase rectifier, a choke, filter condensers, bleeder resistors, an excitation unit for supplying a-c and d-c excitation to the flux gate transmitter and a 400-cycle filter for filtering the output signal from the flux gate transmitter. A phasing resistor for the flux gate signal and a phasing resistor for the gyro signal are contained in the unit. The power transformer and the B-power supply provide all the necessary power for both the flux gate servo amplifier and the directional gyro servo amplifier. The two servo amplifiers are practically identical, consisting of phase discriminating voltage and power amplifier circuits for their particular servo circuits. The mechanism assembly consists of the follow-up autosyns for both the flux gate transmitter and the directional gyro transmitter, the correction and output servomotor rate generator sets, a magnetic clutch, a data transmitting autosyn, a cam compensator with related uncorrected and corrected indicating dials and the necessary interconnecting gearing. The compass coupler is installed adjacent to the electronic control amplifier, AM-608/ARA-25, located in the electrical and radio equipment compartment on the left-hand side of the airplane.

6-150. REMOVING AND INSTALLING COMPASS COUPLER.

REMOVING

1 Remove electrical and radio access panel on left-hand side of airplane.



RADIO AND ELECTRICAL PANEL

2 Cut safety wire and remove two electrical plugs from coupler.

3 Cut safety wire and loosen wing nut fastener on mounting tray until retainer bracket clears lower front surface of coupler.

4 Lift front end of coupler and remove the unit from the airplane.

FJ-48-2-51-105

INSTALLING

1 Position coupler on mounting tray and secure by turning wing nut fastener clockwise until retainer bracket is firm against lower front surface of coupler.

2 With AN995F32 lockwire, safety-wire wing nut to hole provided on front of mounting tray.

3 Connect two electrical plugs to receptacles on front of coupler. Safety-wire connectors with AN995F32 lockwire.

4 Perform an operational check of polar path compass system. (Refer to paragraph 6-140.)

5 When installing a new compass coupler, perform swinging and compensating procedures for the polar path compass system. (Refer to paragraph 6-169.)

6 Replace access door.

FJ-48-2-51-106

6-151. POLAR PATH COMPASS SYSTEM—COMPASS CONTROLLER PANEL.

6-152. The compass controller panel, located at the rear of the right-hand console panel, provides all the manually operated controls and the indications necessary for performing manual operations in the polar path compass system. The panel contains a selector switch ("D.G.," "SLAVED" and "COMP" positions), a voltmeter, a LATITUDE dial which is geared to a potentiometer and a controlling knob, a COURSE SET knob and a synchronizing button (PUSH-SYNC.). The three-position selector switch selects the circuitry in the system for the mode of operation desired. The "D.G." selection only provides directional gyro transmitter references. The "SLAVED" selection allows directional gyro transmitter output signals to be corrected by the output from the flux gate transmitter. The "COMP" selection provides only magnetic references from the flux gate transmitter. The hermetically sealed voltmeter is referred to as a synchronizing indicator, or an annunciator, and its function is to measure the degree of synchronization between the directional gyro information and the flux gate information on the "SLAVED" mode of operation. When the "D.G." mode is selected and the flux gate information is not being used, the synchronizing indicator monitors the drift correction being applied by the LATITUDE dial and potentiometer On the LATITUDE dial, the graduations represent degrees of latitude, clearly marked north (N) or south (S). The latitude of the airplane is set by the operator on this dial by means of a control knob attached to a shaft. Geared to the shaft is a potentiometer. The gearing is adjusted so that movement of the knob selects an output voltage which represents the required correction for the "apparent gyro drift" at the latitude set on the dial. The COURSE SET knob is provided to rapidly change the course heading on the radio magnetic course indicator. The knob is spring-loaded to the out position which frees the shaft of the knob from any gearing. Depressing the knob causes the shaft to engage with gearing to the potentiometer. Turning the knob after depressing it allows an output voltage from the transformer to pass to the appropriate electrical circuit for course setting. This voltage and the speed of the course setting is dependent upon the amount the knob is turned, while the phase of the voltage and the direction the course setting is changed depends upon the direction the knob is turned. If the knob is turned to the right of center ("R"), the radio magnetic course indicator will show an increase in heading. If the knob is turned to the left of center ("L"), the radio magnetic course indicator will show a decrease in heading. When the synchronizing button (PUSH-SYNC.), located below the annunciator on the panel, is depressed, it operates a switch that causes the required circuitry to rapidly synchronize the directional gyro information with the flux gate information.

6-153. REMOVING AND INSTALLING COMPASS CONTROLLER PANEL.

REMOVING

1. Remove four dzus fasteners from the SIF C1272/APA-89 control panel.
2. Lift control panel from console and lay aside.
3. Remove four dzus fasteners from the compass controller panel.

4. Lift controller panel from console.
5. Cut safety wire and remove electrical connector from rear of panel. Cover receptacle with masking tape and tie plug in bag.

FJ-48-2-51-107

6. Remove controller panel from console.

INSTALLING

1. Make electrical connection on rear of controller panel.
2. Safety connector with AN995F32 lockwire.
3. Push wire bundle down and aft in console and position controller panel.
4. Secure controller panel with four dzus fasteners.
5. Position SIF panel in console and secure with four dzus fasteners.
6. Perform an operational check of polar path compass system. (Refer to paragraph 6-140.)

FJ-48-2-51-108

6-154. POLAR PATH COMPASS SYSTEM INDICATIONS.

6-155. Directional gyro reference headings and flux gate transmitter magnetic headings in the polar path compass system are both visually presented on the compass card of the radio magnetic course indicator, ID-250A/ARN, located on the instrument panel. The compass card is calibrated in 360 degrees and is read against a fixed index. Magnetic headings from the flux gate transmitter are indicated on the uncompensated and compensated compass card dials of the compass coupler. These indications are provided in order to check compass deviation error during compass swinging procedures and after compensation.

6-156. POLAR PATH COMPASS SYSTEM SWINGING AND COMPENSATION.

6-157. Compass compensation is the necessary adjustment of the airplane's compass to correct for magnetic deviations. Since the error of any compass is the angular

difference between the true magnetic heading of the airplane and the compass indication, corrections to the compass must be made to enable the pilot to determine the heading of the airplane in relation to the cardinal points of the compass. Compass compensation procedures should be accomplished on the polar path compass system each time a new flux gate transmitter or compass coupler is installed or at any time that the headings are inaccurate on "SLAVED" or "COMP" modes of operation and the trouble cannot be determined. Complete compass swinging and compensation procedures include the following steps:

a. Determining the magnetic heading of the airplane.

b. Swinging the compass on magnetic headings to determine the compass indications.

c. Analyzing steps a. and b. to determine deviations.

d. Compensating the compass to correct for deviations.

e. Determining residual deviations after compensation (calibrating).

f. Completing compass correction cards if required. (No compass correction card is necessary for the polar path compass system.)

6-158. DETERMINING MAGNETIC HEADING OF AIRPLANE.

6-159. In order to determine the deviation necessary to compensate the compass and to record residual values, it is first necessary to align the airplane on the magnetic heading. This may be accomplished by either the compass rose or the sighting compass method. The surveyor's transit may also be used.

6-160. COMPASS ROSE METHOD. The compass rose is a circular area permanently marked off in degrees from magnetic North. The compass rose method of alignment consists of placing the airplane on the compass rose so that its longitudinal axis is parallel to a radial bar which can be positioned and locked in predetermined magnetic directions. The airplane rests on a pivot so that it can be swung by hand to any desired magnetic heading.

Note

The magnetic markings on the compass rose must have been checked by a licensed surveyor within one year previous to the date of swinging of any airplane.

6-161. SIGHTING COMPASS METHOD. The sighting compass method requires the use of a standard compass which has had its compensating assembly replaced by a swinging compass sight. To determine the magnetic heading of the airplane, stand at least 50 feet behind the airplane and, with the swinging compass in cupped hands, align the sight with the longitudinal axis of the airplane. Refer to a row of fuselage rivets or any suitable line or pair of objects on the airplane. When the sight hairline and the objects sighted are aligned, the reading on the sighting compass will indicate the magnetic headings of the airplane. If the reading is taken in front of the airplane, obtain the magnetic heading by adding or subtracting 180 degrees.

6-162. PRESWINGING AND PRECOMPENSATION PRECAUTIONS.

6-163. To obtain correct compass deviations and, subsequently, make accurate corrections to the compass, certain precautions must be taken:

a. Personnel engaged in compensation of compasses must remove all magnetic materials from their person such as pocket knives, tools, mechanical pencils, etc.

b. All equipment in the airplane having any magnetic effect on the compass must be secured in the position occupied in normal flight.

c. Any magnetic objects such as tractors, automobiles or other airplanes must be removed from the swinging area to a point at which they will have no effect on the compass.

d. Any equipment necessary to compensate the compass, such as a battery cart, must be moved in a circle about the stationary airplane in order to test the magnetic effect of the equipment on the compass. The equipment must be in the same position and at the same distance from the airplane as it would be used in the swinging of the compass. The changes in compass reading due to such movement shall not exceed ¼ degree.

6-164. INTRODUCTION TO COMPENSATION OF POLAR PATH COMPASS SYSTEM.

6-165. All constant magnetic deviation errors in the polar path compass system are corrected at the compass coupler unit. (See figure 6-37.) (All index error due to faulty mounting is corrected by realigning the flux gate transmitter.) (Refer to paragraph 6-166.) The compensating mechanism in the compass coupler, which contains no magnets, is entirely mechanical in operation and the deviation corrections are made on the compensated dial by the adjustment of 12 screws arranged in a circle and spaced at 30-degree intervals with reference to North. Access to the adjusting screws is gained by removing the dust cover on the front of the compass coupler. The back ends of these screws are attached to a flat, circular metal cam over which rides a cam follower wheel. Turning the compensating adjusting screws to the right or left changes the contour of the metal cam, thus causing the cam follower to rise or fall as it passes over that section. The cam follower is attached to a correction mechanism which introduces a spread between the readings of the compensated dial and the uncompensated dial. The compensated dial can thus be made to lead or lag the uncompensated dial at those points where adjustments are to be made. Both the uncompensated and the compensated dials are seen through a window located on top of the compass coupler.

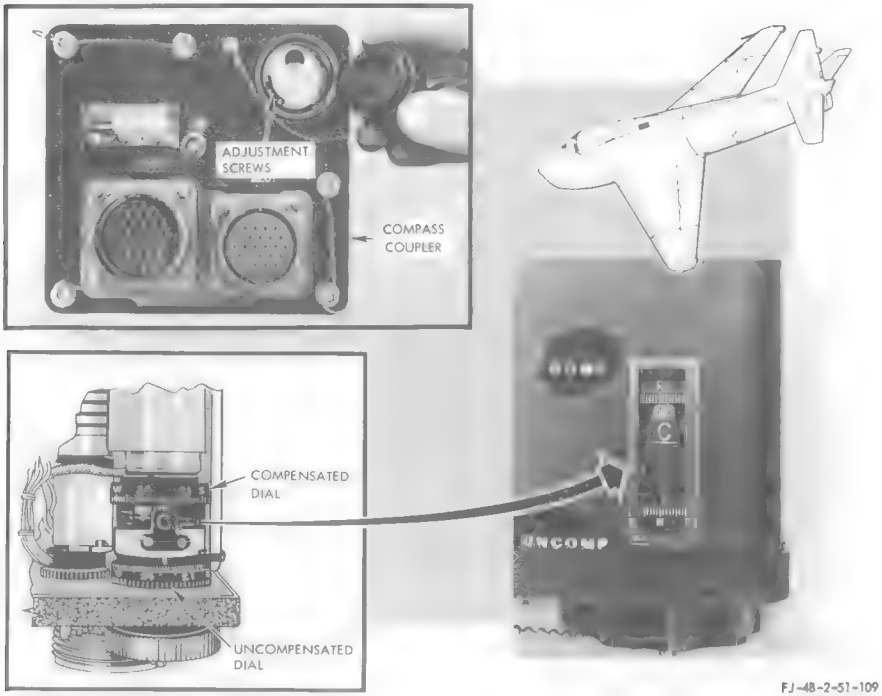
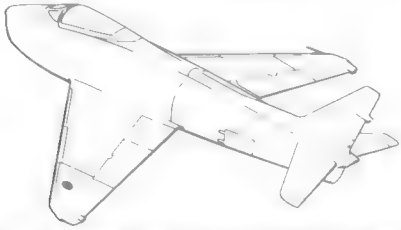


Figure No. 6-37. Polar Path Compass System Compensation Adjustments

6-166. REALIGNING FLUX GATE TRANSMITTER TO COMPENSATE FOR INDEX ERROR.



Note If the index error is one degree or greater, the flux gate transmitter must be realigned. Proceed as follows:

1. Remove remote compass access panel on top of left outboard wing.
2. Loosen the three mounting screws on the transmitter unit.

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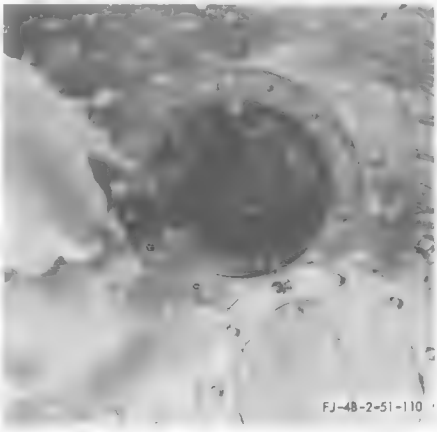
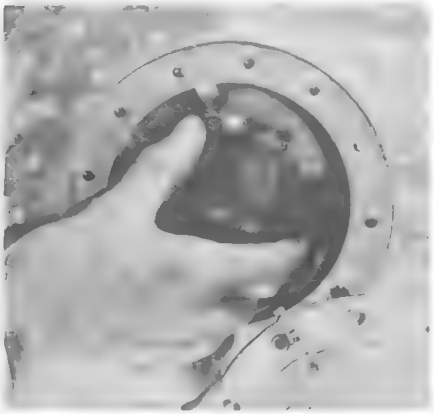


3 If the index error is positive, rotate the transmitter counterclockwise in its mounting. If the index error is negative, rotate the transmitter clockwise.



Note Rotate transmitter correct distance by utilizing the index marks on the graduated scale which is calibrated in one-degree increments. The transmitter should be rotated until the uncompensated dial on the compass coupler indicates the magnetic heading of the airplane plus or minus this alignment error.

4 Carefully holding the transmitter in place, tighten the three mounting screws securely.

5 Replace access panel.

FJ-48-2-51-111

6-167. COMPENSATION PRECAUTIONS.

6-168. In order to make correct adjustments to the compensated dial, the following precautions should be observed:

a. Be sure to use the proper size Allen wrench when adjusting the compensating screws.

b. Adjust all compensating screws in consecutive order; do not skip from heading to heading. It is not necessary to try to complete the compensation procedure with a single adjustment of each screw. Better results will be obtained by going around the circle of adjusting screws several times.

c. Do not attempt to take out more than 2 degrees of deviation at the first adjustment of any screw. If deviation at any point exceeds 2 degrees, return to that point after the other screws have been adjusted and take out the remainder.

d. Do not attempt to make a compensating adjustment which exceeds by more than 2 degrees the adjustment made at the preceding screw. To do so may buckle the compensating cam. Investigate to determine the cause of the excessive deviation error change. For example, if the deviation error on a heading of 120 degrees is +1, and the deviation error on the next heading of 150 degrees is +4 degrees (or a spread of 3), do not adjust but investigate the cause of the error on the 150-degree heading.

6-169. PROCEDURE FOR SWINGING AND COMPENSATING POLAR PATH COMPASS SYSTEM.

6-170. To accomplish ground swinging and compensating procedures for the polar path compass system, proceed as follows:

a. Check to see that all components are installed correctly.

Note

To aid in compensating the compass system, the compass coupler may be removed from its normal location in the airplane (paragraph 6-150) and moved to the cockpit area. A pair of extension cables will be needed.

b. Place the d-c power switch, located on the right-hand console, in the "OFF" position.

c. Connect a source of external power to the airplane and select the "SLAVED" mode of operation by means of the selector switch. Allow the system to warm up for at least 3 minutes before taking any readings.

d. Align the airplane on the four cardinal headings, using the desired procedure, and read the uncompensated dial at the compass coupler. (See figure 6-37.)

Note

- The uncompensated and the compensated dials at the compass coupler are read during the index swing and the compensating swing. During the final or residual swing, the readings are taken on the compass card of the radio magnetic course indicator.
- For each heading of the airplane, the annunciator meter on the compass controller panel should be checked for a null reading before reading the uncompensated dial (or allow 2 minutes to elapse after each alignment). Otherwise, the compass would be swung as a directional gyro rather than a magnetic slaved directional gyro. A 1/16-inch movement of the annunciator needle is equal to approximately one degree of misalignment between the flux gate transmitter and the directional gyro transmitter. The PUSH-SYNC. knob may be pushed to remove any misalignment.

e. Subtract each uncompensated dial reading from the magnetic heading. For example, if the magnetic heading is 90 degrees and the uncompensated dial reading is 93, the deviation error is −3 degrees. Average the four cardinal heading deviation errors by adding them together and dividing by 4. The final value is the index error.

f. If the index error is one degree or greater, the flux gate transmitter must be realigned. (Refer to paragraph 6-166.)

g. If it was necessary to realign the transmitter, repeat the index swing, again figuring the deviation error on the four cardinal headings. If no deviation error in excess of 2 degrees is found on any of the cardinal headings after the index error has been removed, proceed to step h.

h. Align or swing the airplane to 12 magnetic headings, 30 degrees apart, starting at North (000). On each heading read and record the uncompensated dial reading versus the actual airplane magnetic heading and find the deviation value. Press and hold down the PUSH-SYNC. knob and adjust the correct compensating screw (center screw of three visible screws) until the compensated dial reads the correct magnetic heading.

Note

Whenever making an adjustment to a compensating screw, press and hold down the PUSH-SYNC. knob. Any movement of the compensating screw will directly move the uncompensated dial. In order to move the compensated dial, this correction signal must pass through the compass coupler servo mechanism circuits (normally on slow slave). Therefore, the system must be maintained in fast slave to cause any movement of the compensating screws to move the compensated dial.

Release the PUSH-SYNC. knob. Check the reading of the compensated dial and the compass card of the radio magnetic course indicator. The two should agree with each other within 0.5 degree. Proceed to the next heading until all compensating screws have been adjusted.

i. If deviation in excess of 2 degrees is found on any of the cardinal headings after any index error has been removed, proceed to steps j. through o.

j. Align or swing the airplane to 12 magnetic headings, 30 degrees apart, starting at North (000). Record the magnetic heading and the corresponding reading of the uncompensated dial of the compass coupler. (See figure 6-37.)

Note

If the compass rose method of determining the magnetic heading is used, corrections to the magnetic meridians due to variation may result in the "cardinal" markings being changed by a few degrees. Alignment with these meridians will be more accurate than attempting to use true N, S, E or W.

k. After the airplane has been swung to the 12 headings, 30 degrees apart, calculate and tabulate the deviation, with its sign, at each heading by subtracting the uncompensated dial reading from the magnetic heading. (See figure 6-38.) Plot the deviation curve which is the deviation against the actual magnetic heading at each point of reading. This deviation curve should be a sine wave. If it is not, the system is defective.

l. Place the airplane at the magnetic heading corresponding to the adjustment point at which the deviation error is a minimum. In the illustrated example (figure 6-38), this point is 150 degrees. Read the compensated dial of the compass coupler. The dial reading should be approximately the same as the magnetic heading since this is the point of minimum deviation.

m. Press and hold down the PUSH-SYNC. knob, located on the compass controller panel, and adjust the correct compensating screw (center one of the three that are visible) until the compensated dial reads the correct airplane heading. Release the PUSH-SYNC. knob.

n. Check the reading of the compensated dial and the radio magnetic course indicator. The two should agree with each other within 0.5 degree.

o. Proceed to the next heading and repeat steps m. and n. until all 12 compensating screws have been adjusted.

Note

Be sure to limit the adjustments as noted in the compensation precautions (paragraph 6-167). If necessary, repeat the complete procedure to get the compensated dial reading closer to the magnetic heading at each of the 12 adjustment points. There should be no noticeable deviation at any heading.

p. When satisfied that the least possible deviation exists at each point of adjustment, reinstall the compass coupler. (Refer to paragraph 6-150.)

q. To check for residual deviation, align or swing the airplane on all 12 headings, 30 degrees apart, starting at North. On each heading read and record the heading indicated by the compass card of the radio magnetic course indicator. Subtract each of these indications from the true magnetic heading at that test point. If the residual deviation meets with the requirements of the compensated deviation spread (paragraph 6-171), the compass compensation is considered complete. If it does not, perform an operational check of the system (paragraph 6-140) and repeat the entire swinging and compensating procedure. No compass correction card is necessary for the polar path compass system.

6-171. COMPASS DEVIATION REQUIREMENTS—POLAR PATH COMPASS SYSTEM.

6-172. The difference between the maximum positive deviation and the maximum negative deviation for the same swinging procedure is referred to as the maximum "spread." In the illustrated example (figure 6-38), note that the maximum positive deviation (uncompensated)

1	2	3
ACCURATE MAGNETIC HEADING (OBTAINED IN SHIP SWINGING)	UNCORRECTED DIAL HEADING CORRESPONDING TO ACCURATE MAGNETIC HEADING	DEVIATION
3	4	− 1
32	35-1/2	− 3-1/2
56-1/2	61-1/4	− 3-3/4
90	93	− 3
121	122-3/4	− 1-3/4
149	149-1/8	− 1/8
182	180-1/4	+ 1-3/4
210	206-3/4	+ 3-1/4
244	240	+ 4
271	267-1/2	+ 3-1/2
302	299-1/2	+ 2-1/2
328	327	+ 1
351-1/2	351-1/2	0

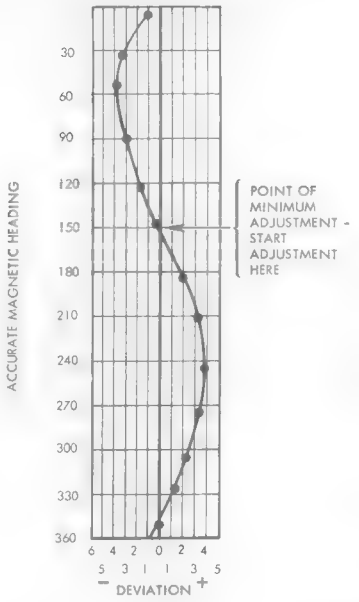


Figure No. 6-38. Compass Deviation Curve

is +4 and that the maximum negative deviation is −3¾. The "spread" is 7¾. If the maximum "spread" for the uncompensated swing exceeds these limits, the system is defective and should not be compensated. Determine the deviation "spread" for the residual (compensated) swing. (Refer to step q., paragraph 6-170.) The allowable maximum "spreads" for the polar path compass system are:

a. *Uncompensated deviation:* the maximum uncompensated "spread" should not exceed 8 degrees.

b. *Compensated deviation:* the maximum compensated "spread" should not exceed 2 degrees. The maximum deviation on any one heading should not exceed one degree.

6-173. STAND-BY COMPASS.

6-174. A direct-reading, magnetic stand-by compass (figure 6-39) is installed on the top left-hand side of the instrument panel shroud. A name plate, located adjacent to the compass, reads "STAND-BY COMPASS EMERGENCY USE ONLY." Within the compass is a metal bowl filled with compass fluid and a semi-float type compass card. A pair of magnets, attached to the compass card, align with the earth's magnetic field to present indications. The compass card is graduated in 5-degree increments through 360 degrees and has the cardinal heading indicated by "N," "E," "S" and "W." A compensating system, utilizing built-in permanent magnet compensators, is attached to the compass for the purpose of correcting deviations which result from magnetic disturbances. This compass should be compensated every 100 hours or at least once in each 3-month period and at times when a change of equipment is made that might affect the instrument. The compass is illuminated by 28-volt d-c integral lighting and receives power from the primary bus. A separate on-off switch (STANDBY COMPASS & RANGE IND) and an instrument lights rheostat (INSTRUMENTS), both located on the right-hand console panel, control the lighting circuit to the stand-by compass.

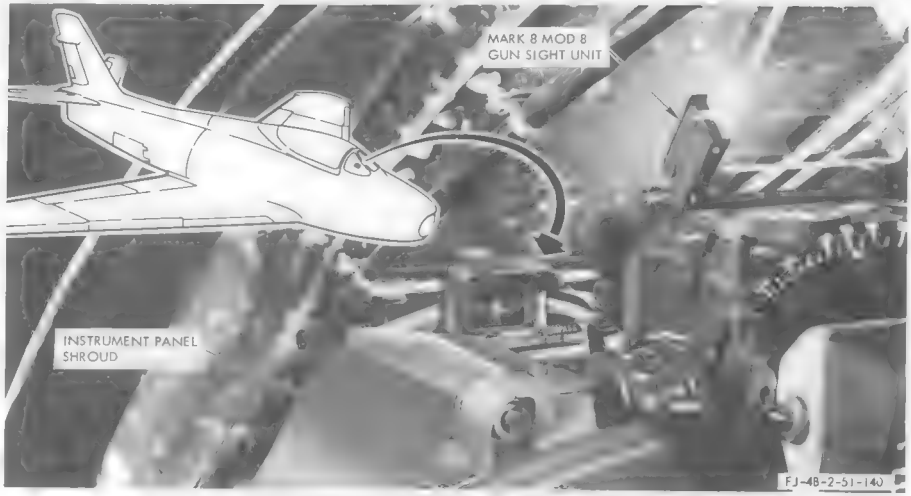


Figure No. 6-39. Stand-by Compass

6-175. PREVENTIVE MAINTENANCE OF STAND-BY COMPASS.

a. Check the compass for dirty, broken or loose cover glass.

b. Inspect the compass visually for discoloration of liquid and for evidence of bubbles. Clouded or discolored liquid will impair visibility.

c. Check compass for tightness of screws to eliminate leakage of compass liquid. If loss of liquid is obvious, replace compass since the card assembly will not be damped sufficiently, resulting in erroneous indications.

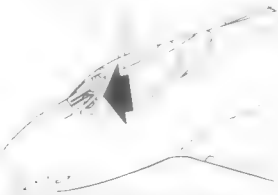
d. Check the compass correction card to see that the compass has been compensated within 100 hours (or within a 3-month period).

6-176. TROUBLE SHOOTING STAND-BY COMPASS.

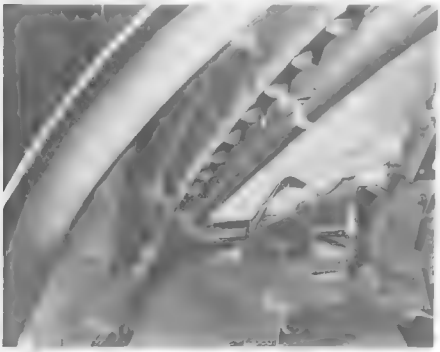
PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
DISCOLORED LIQUID.		
Deterioration within the compass.		Replace compass. (Refer to paragraph 6-177.)
FADING OR ILLEGIBILITY OF CARD MARKINGS.		
Deterioration within the compass.		Replace compass. (Refer to paragraph 6-177.)
CARD DOES NOT ROTATE FREELY WHEN AIRPLANE IS IN NORMAL FLYING POSITION. SLUGGISH MOVEMENT.		
Broken pivot; broken or cracked jewel.		Replace compass. (Refer to paragraph 6-177.)
COMPASS IS ERRATIC OR DOES NOT RESPOND TO COMPENSATION.		
Magnets demagnetized or compensating gear magnets demagnetized.		Replace compass. (Refer to paragraph 6-177.)
LOSS OF COMPASS LIQUID.		
Damaged filling screw gasket. Window gasket damaged. Bowl cracked.		Replace compass. (Refer to paragraph 6-177.)

6-177. REMOVING AND INSTALLING STAND-BY COMPASS.

REMOVING



1 Remove top two instrument mounting screws that secure the visor to the compass mounting bracket and remove visor.



2 Remove bottom two mounting screws.

3 Push the compass forward to clear the mounting bracket and pull to the right of bracket.

4 Remove electrical plug from compass receptacle.

5 Remove compass from the airplane.

FJ-48-2-51-114

INSTALLING

1 Perform preventive maintenance of stand-by compass. (Refer to paragraph 6-175.)

2 Hold the compass forward of the mounting bracket and make electrical connection.

3 Position the compass in the bracket and push the face of the dial aft against the cutout.

4 Position the visor at the top aft side of the mounting bracket and secure with top two instrument mounting screws.

5 Install bottom two instrument mounting screws.

FJ-48-2-51-115

6-178. TESTING AND REPAIRING STAND-BY COMPASS FOR LEAKAGE.

TESTING

1 Place compass in a bell jar.

2 Using a source of suction, evacuate the bell jar to a pressure equivalent to an altitude of 50,000 feet.

3 Allow the compass to remain in the bell jar for one-half hour.

4 Observe compass for signs of leaking fluid through the light lens. If leakage occurs, repair compass.

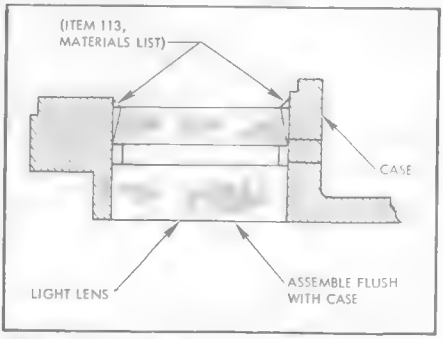
REPAIRING

1 Disassemble the compass to the extent that the light lens is accessible.

3 Make an applicator from a piece of music wire with a small loop at the end.

2 Clean the surfaces of the light lens with trichlorethylene (item 134, materials list) to remove dust, oil, etc.

4 Dip applicator into sealing compound (item 113, materials list) and apply a thin coating in the corner formed by the lens and the compass housing. Allow to dry for 15 minutes.



5 Apply a second coat and allow to air-dry for at least 24 hours.

FJ-48-2-51-112

Note The top surface of the light lens is frosted. Keep the sealing compound away from the center of the lens. If some sealing compound should accidentally fall on the center of the lens, it will not have any adverse effect on the lighting characteristics. However, it should be avoided if possible.

6 Fill chamber of compass with clean compass fluid (item 53, materials list). Dip unit in trichlorethylene (item 134, materials list) to remove the compass fluid from the external surfaces. Thoroughly dry the external surfaces.

7 Set the compass on a clean blotter and place both compass and blotter in a bell jar. Reduce the pressure in the bell jar equivalent to an altitude of 50,000 feet. Hold the pressure for one-half hour. After this time, increase the pressure to ambient and, if existent, examine the stain on the blotter.

8 If there is an apparent leak, replace the compass. Remove the defective unit to an instrument shop for complete disassembly and repair.

FJ-48-2-51-113

6-179. COMPASS CORRECTION CARD.

6-180. A compass correction card (figure 6-40) is provided for the stand-by compass as an aid to navigation. The card is located in a card holder which is mounted on the canopy to the right of the instrument panel.

6-181. STAND-BY COMPASS SWINGING AND COMPENSATION.

6-182. The stand-by compass must be swung to determine the amount of deviation between the compass indications and the true magnetic heading. Adjustments must then be made to the compass to compensate for this error. The coefficient method of compass swinging and compensating is used for the stand-by compass. This method permits deviations to be analyzed and corrected with the least amount of error. After deviations have been determined on the four cardinal headings, coefficients, which are determined *algebraically*, are used in applying the proper amount of correction to the compass. The three coefficients used in this procedure are designated "A," "B" and "C," but the stand-by compass does not need to be compensated for coefficient "A."

6-183. CALCULATING COEFFICIENT "B." Coefficient "B" is caused by the fore-and-aft horizontal component of the airplane's hard iron. This value is calculated after the airplane has been swung on an East-West heading and is the deviation on East minus, *algebraically*, the deviation on West divided by 2.

$$\frac{\text{(DEV. on E.)} - \text{(DEV. on W.)}}{2}$$

6-184. CORRECTING FOR COEFFICIENT "B." To correct for coefficient "B," add, *algebraically*, the value of "B" to the compass reading with the airplane headed magnetic East; make the compass indicate the corrected reading by adjusting the E-W compensating screw.

6-185. CALCULATING COEFFICIENT "C." Coefficient "C" is the amount of correction required to compensate for the lateral horizontal component of the airplane's hard iron. This value is calculated after the airplane has been swung on a North-South heading

	COMPENSATING SWING			RESIDUAL SWING	
	ACTUAL HEAD (M)	AIRCRAFT COMP.	DEV'N	ACTUAL HEAD (M)	AIRCRAFT COMP.
NO 000	000	356	+ 4	000	001
NE 045				045	045
E 090	090	093	-3	090	090
SE 135				135	134
S 180	180	184	-4	180	179
SW 225				225	224
W 270	270	268	+ 2	270	270
NW 315				315	315
	(1)	(2)	(1) - (2)	(3)	(4)

IF SWINGING COMPASS USED AHEAD OF AIRCRAFT, ADD OR SUBRACT 180 DEGREES.

COEF $C = \frac{N - S}{2} = \frac{(+4)-(-4)}{2} = +4$

COEF $B = \frac{E - W}{2} = \frac{(-3)-(+2)}{2} = -2\text{-}1/2$

COEF $A = \frac{N + E + S + W}{4} = \frac{(+4)+(-3)+(-4)+(+2)}{4} = -1/4$

COMPASS SWUNG _____ BY: _____

TO FLY	STEER	TO FLY	STEER
N	001	180	179
15	015	195	194
30	031	210	209
45	045	225	224
60	060	240	239
75	075	255	254
90	090	270	270
105	105	285	286
120	120	300	300
135	134	315	315
150	149	330	331
165	164	345	346

FJ-48-2-51-151

Figure No. 6-40. Compass Correction Card

and is the deviation on North minus, *algebraically*, the deviation on South divided by 2.

$$\frac{\text{(DEV. on N.)} - \text{(DEV. on S.)}}{2}$$

6-186. CORRECTING FOR COEFFICIENT "C." To correct for coefficient "C," add, *algebraically*, the value of "C" to the compass reading with the airplane headed magnetic North; make the compass indicate the corrected reading by adjusting the N-S compensating screw.

6-187. PROCEDURE FOR SWINGING AND COMPENSATING STAND-BY COMPASS.

6-188. To accomplish ground swinging and to compensate the pilot's stand-by compass, proceed as follows:

Note

Observe preswinging and precompensation precautions as outlined in paragraph 6-162.

a. Head the airplane toward magnetic South, using the desired aligning procedure. (Refer to paragraphs 6-158 through 6-161.)

Note

All headings must be exact magnetic headings.

b. Level the airplane so that the compass is horizontal within 5 degrees.

c. Set the compensators to null effect by turning both E-W and N-S compensating screws until the dots on the screws are matched with the dots on the compass.

d. Note the compass reading on the magnetic South heading; determine the compass deviation by subtracting, *algebraically*, the compass reading from the exact magnetic heading of the airplane. Record the deviation on the compass correction card (figure 6-40) for the compensating swing.

e. Head airplane toward magnetic West and note compass reading. Determine the compass deviation on this heading and enter information on compass card.

f. Head the airplane toward magnetic North and note the compass reading. Determine the compass deviation and record on the compass card. Calculate coefficient "C" and correct for the value of coefficient "C."

g. Head the airplane to magnetic East and note the compass reading. Determine the compass deviation and record on the compass card. Calculate coefficient "B" and correct for the value of "B."

6-189. FILLING OUT COMPASS CORRECTION CARD.

6-190. To fill out the compass correction card (figure 6-40) for the stand-by compass, proceed as follows:

a. Enter the magnetic South heading of the airplane in the first column under the appropriate label "ACTUAL HEAD (M)."

b. Enter the compass South heading in the second column under "AIRCRAFT COMP."

c. Determine the deviation by subtracting column 2 from column 1 and record this value in column 3 labeled "DEV'N."

Note

For reference purposes, the subtraction is indicated at the bottom of the column, (1) − (2).

d. Repeat this procedure on the other three cardinal headings, West, North and East in that order. This procedure will fill in the first three columns.

e. Using the deviations in column 3, calculate, *algebraically*, the coefficients "B" and "C."

f. Compensate the compass using the coefficients thus calculated.

Note

Steps a. through f. will have been accomplished during the compensating swing procedure.

g. Swing the airplane on the headings indicated on the compass correction card in the column labeled "TO FLY." Start this swing from the last heading upon which the compass was compensated. These headings must be exact magnetic headings. At each magnetic heading, record the compass reading in the column labeled "STEER."

h. Complete the correction card by filling in columns (3) and (4) under "RESIDUAL SWING" using eight headings.

i. Check the compass deviation requirements (paragraph 6-191) to see that the compensated deviations do not exceed the allowable maximum "spread."

j. Supply the information required on the reverse side of the card.

k. Insert the smaller part of the card in the compass correction card holder in the airplane. The remainder of the card is to be filed with the airplane inspection sheets.

6-191. COMPASS DEVIATION REQUIREMENTS—STAND-BY COMPASS.

6-192. The maximum "spread" (the arithmetic difference between the maximum positive deviation and the maximum negative deviation) which is allowed for the compensating swing of the stand-by compass is 28 degrees. If this limit is exceeded, do not attempt to compensate the compass; replace it with a known good compass. If the uncompensated deviation "spread" is less than 28 degrees and the compass is then compensated, the allowable maximum "spread" during the residual swing should not exceed 10 degrees. If this limit is exceeded, repeat the swinging and compensating procedure.

ANGLE-OF-ATTACK AND RELATED SYSTEMS

6-193. ANGLE-OF-ATTACK AND RELATED SYSTEMS.

6-194. The angle-of-attack and related systems consist of the angle-of-attack indicating system, the rudder pedal shaker system, the approach light system and, on airplanes 143594m and subsequent, the approach indexer system. The angle-of-attack indicating system detects the local angle of attack of the airplane from a point on the side of the fuselage and furnishes reference information for the control and actuation of other units and systems in the airplane (figures 6-41 and 6-42). Signals from the angle-of-attack airstream direction detector are provided to operate an angle-of-attack indicator, located on the instrument panel, where a continuous visual indication of the local angle of attack is displayed. The angle-of-attack indicator contains switches that are actuated at preset angles of attack to energize the red, amber and green lights in the approach light system. These lights automatically furnish the landing signal officer with an accurate indication of the landing approach angle of attack. On airplanes 143594m and subsequent, electrical signals from the angle-of-attack indicator are also sent to the approach indexer which provides the pilot with an illuminated indication of the landing approach angle of attack. The angle-of-attack indicator also provides electrical signals to operate the rudder pedal shaker which warns the pilot when the airplane is approaching the critical stall angle of attack. The angle-of-attack and angle-of-sideslip compensator receives local angle-of-attack and local angle-of-sideslip signals from the airstream direction detectors and functions of Mach from the pressure ratio transducer. This information is converted to true angle-of-attack and true angle-of-sideslip signals which are then supplied to the fire control system for controlled rocket firing.

6-195. ANGLE-OF-ATTACK INDICATING SYSTEM.

6-196. The angle-of-attack indicating system consists of an airstream direction detector, located on the left side of the forward fuselage, and an indicator located on the pilot's instrument panel. The airstream direction detector contains the sensing element which measures local airflow direction relative to the true angle of attack by determining the angular difference between local airflow and the fuselage reference plane. The sensing element operates in conjunction with a servo-driven balanced bridge circuit which changes probe position to electrical signals. Operating power for the angle-of-attack system is 28 volts dc from the primary bus. The system is protected by a 5-ampere circuit breaker (ANGLE OF ATTACK IND) located on the left-hand radio bay circuit-breaker panel.

CAUTION

Do not use the angle-of-attack airstream direction detector as a step or handgrip. This could easily result in bending the probe, causing it to stick or bind and rendering the system inoperable or inaccurate.

Note

A cover is provided for the probe of the angle-of-attack airstream direction detector. This cover should be installed at all times when the airplane is on the ground to keep moisture and dirt from entering the probe.

6-197. FUNCTION OF ANGLE-OF-ATTACK INDICATING SYSTEM—PHYSICAL.

6-198. The operation of the angle-of-attack indicating system is based on the detection of differential pressure which is caused by changes in airflow around the probe unit. The probe extends through the skin of the airplane into the air stream. At the location of the airstream direction detector on the left side of the fuselage, the ratio of angular change of local airflow direction to change of true angle of attack is approximately 1.7 to 1.0. The exposed end of the probe contains two parallel slots which detect the differential airflow pressure. Air from the slots is transmitted through two separate air passages to separate compartments in a paddle chamber. Any differential pressure, caused by misalignment of the probe with respect to the direction of airflow, will cause the paddles to rotate. The moving paddles will rotate the probe, through mechanical linkage, until the pressure differential is zero. This occurs when the slots are symmetrical with the air stream direction.

6-199. FUNCTION OF ANGLE-OF-ATTACK INDICATING SYSTEM—ELECTRICAL.

6-200. Two electrically separate potentiometer wipers, rotating with the probe, provide signals for the angle-of-attack indicator and the angle-of-attack and angle-of-sideslip compensator. Probe position, or rotation, is converted into an electrical signal by one of the potentiometers which is the transmitter component of a self-balancing bridge circuit. When the angle of attack of the airplane is changed and the position of the transmitter potentiometer is altered, an error voltage exists between the transmitter potentiometer and the receiver potentiometer in the indicator. This error voltage and polarity is detected and amplified to drive a servomotor in the correct direction to reposition the receiver potentiometer to provide a null or electrically balanced condition. The

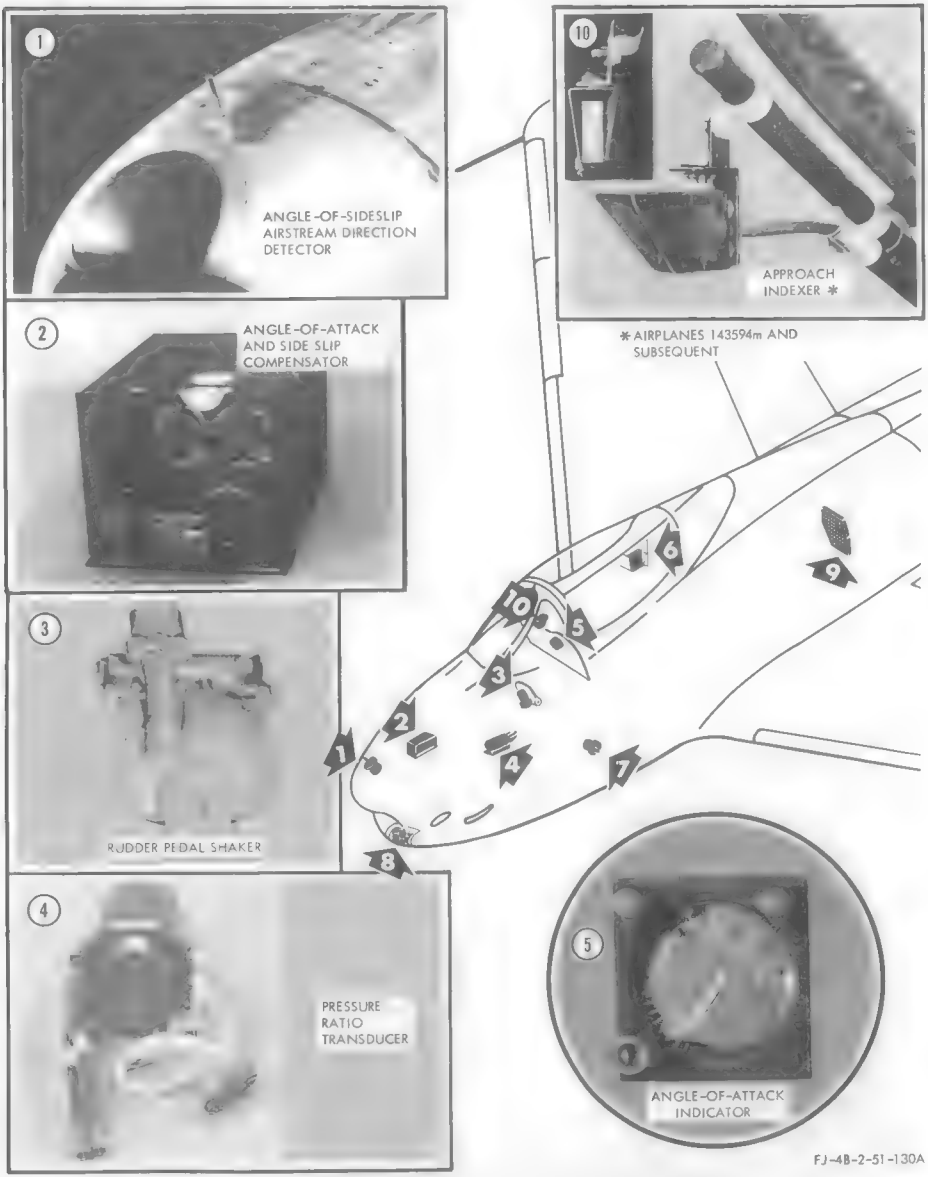


Figure No. 6-41. Angle-of-Attack and Related Systems (Sheet 1)

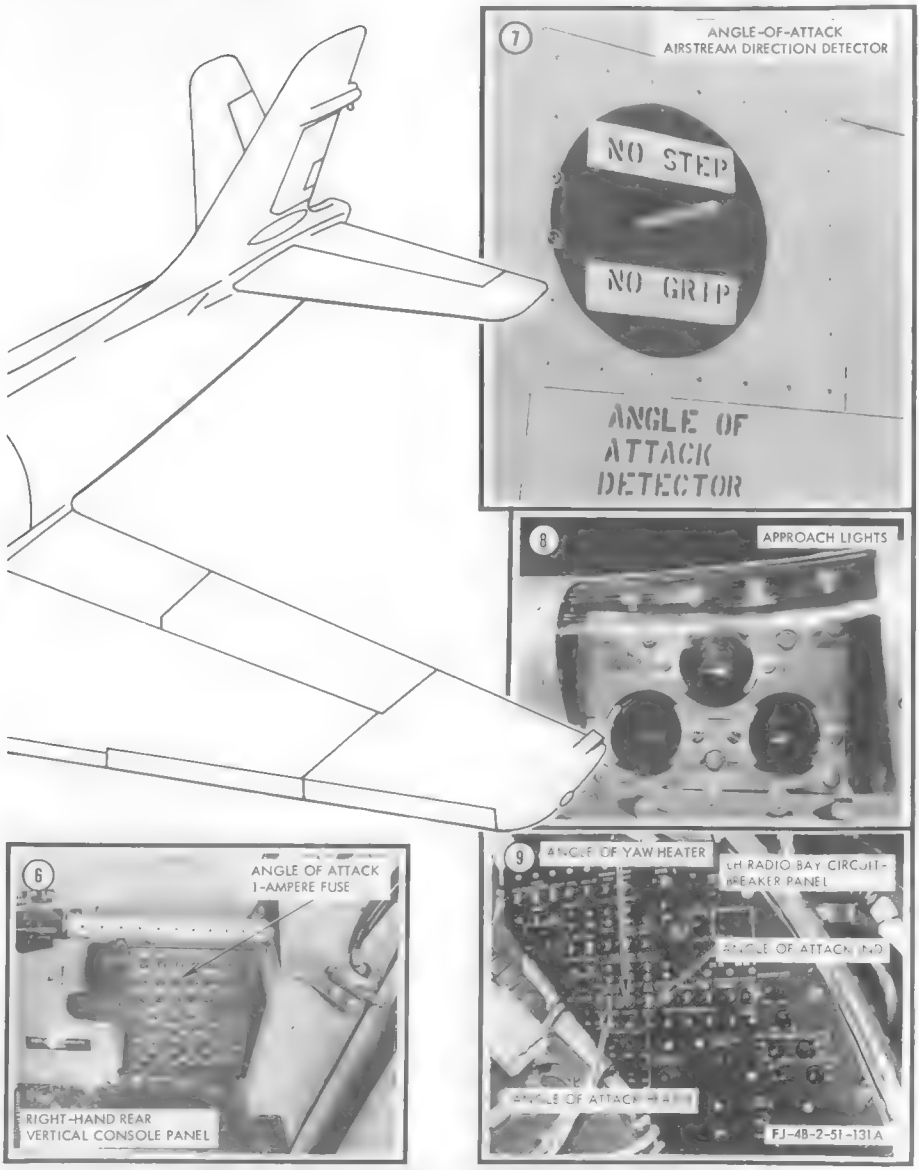


Figure No. 6-41. Angle-of-Attack and Related Systems (Sheet 2)

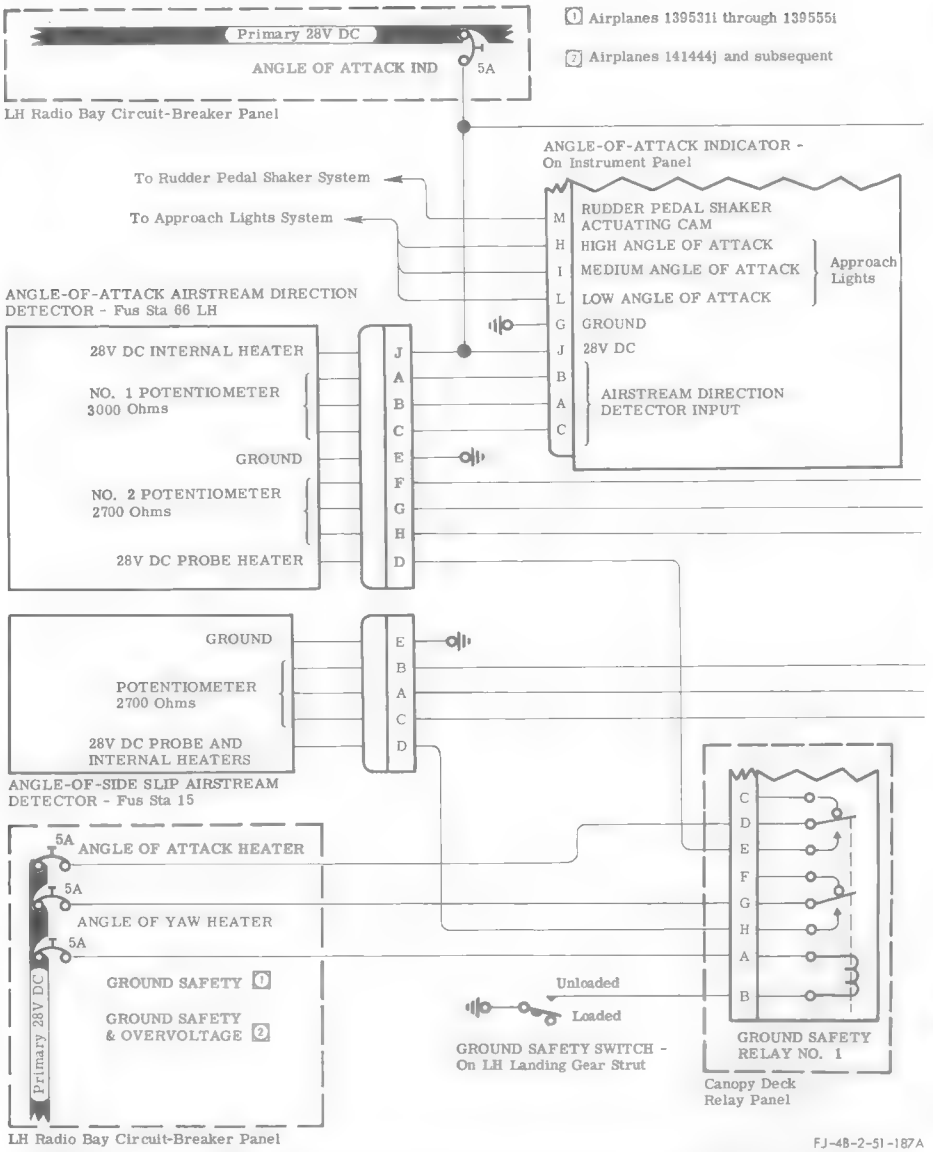


Figure No. 6-42. Angle-of-Attack and Angle-of-Sideslip System Schematic (Sheet 1)

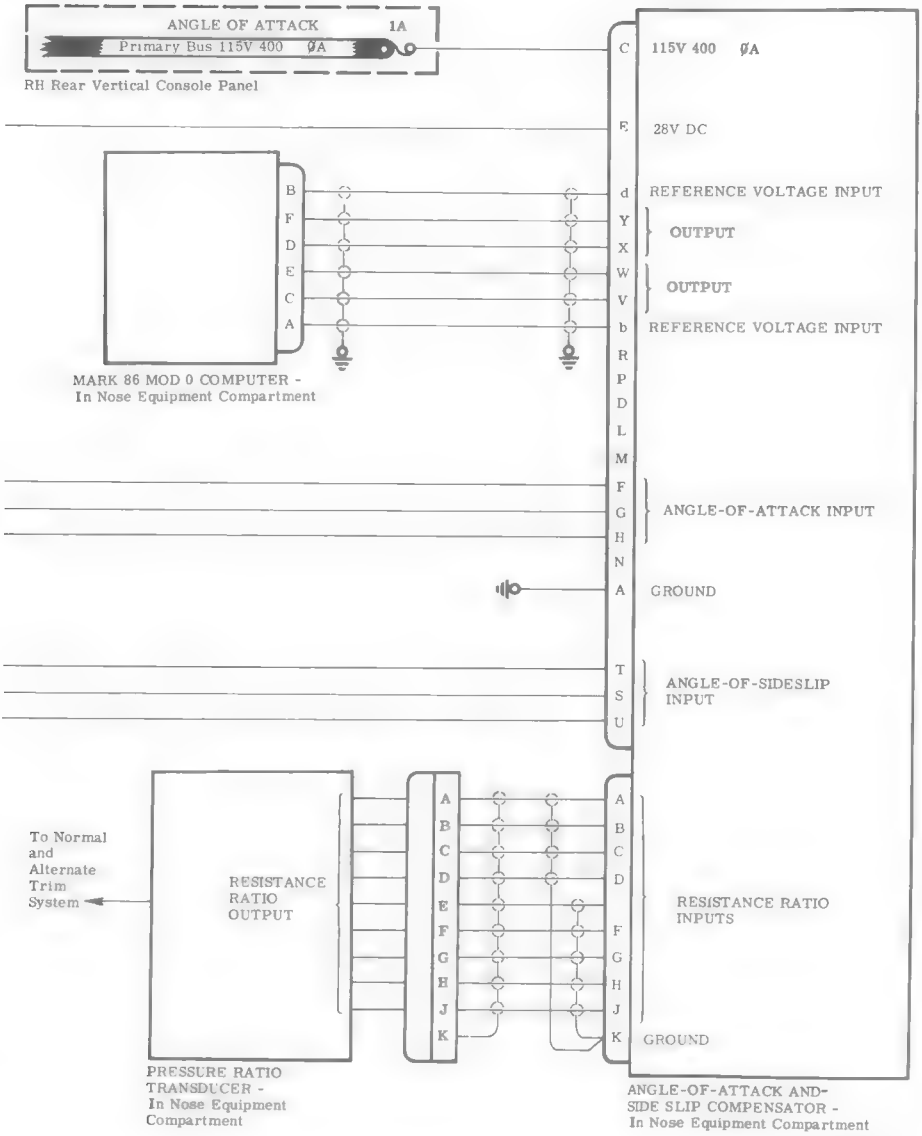


Figure No. 6-42. Angle-of-Attack and Angle-of-Sideslip System Schematic (Sheet 2)

polarity of the error voltage determines the direction of rotation of the servomotor. The indicating pointer moves with the receiver potentiometer wiper arm to indicate on the dial the relative angle of attack. The error voltage is fed back to the error voltage detector and becomes of the opposite polarity. This voltage then opposes the error voltage. When the receiver potentiometer is being driven to a position to balance the circuit, the error voltage is decreasing and the reverse polarity feedback opposes the error voltage. As the system approaches electrical balance, there follows a period of alternating reversals of voltage polarity. These voltages add together algebraically so that even periods of oscillation will occur about electrical zero when the system is in balance. This intentional oscillation eliminates oscillation or hunting of the indicating pointer.

6-201. TROUBLE SHOOTING ANGLE-OF-ATTACK INDICATING SYSTEM.

TEST EQUIPMENT: D-C voltmeter.
A-C voltmeter.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND circuit breaker engaged.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
INDICATOR POINTER MOVEMENT DOES NOT CORRESPOND WITH AIRSTREAM DIRECTION DETECTOR PROBE MOVEMENT.			
No d-c power to airstream direction detector.	Check between test point FAA and ground.	28 volts dc.	Replace defective airstream direction detector. (Refer to paragraph 6-211.)
		Zero volts.	Replace defective d-c power wire(s).
No d-c power to angle-of-attack indicator.	Check between test point FAB and ground.	28 volts dc.	Replace defective angle-of-attack indicator. (Refer to paragraphs 6-7 and 6-8.)
		Zero volts.	Replace defective d-c power wire(s).
Airstream direction detector probe bent or damaged.	Blow into each slot through a short piece of tubing. (Do not blow into both slots at once.) Probe should rotate in the air stream.	None.	Replace airstream direction detector. (Refer to paragraph 6-211.)
Airstream direction detector improperly installed.	Visually inspect airstream direction detector installation making certain that the two dowel pins on the detector are mated with the holes in the mounting plate.	None.	Remove airstream direction detector and install per instructions in paragraph 6-211.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK HEATER and GROUND SAFETY* (GROUND SAFETY & OVERVOLTAGE†) circuit breakers engaged.
Airplane on jacks or ground safety switch linkage disconnected and moved to the unloaded position. (Refer to paragraph 6-203.)

CAUTION

Do not allow angle-of-attack probe to overheat.

*Airplanes 139531i through 139555i
†Airplanes 141444j and subsequent

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
ANGLE-OF-ATTACK PROBE HEATER INOPERATIVE.			
Heater element defective.	Check between test point FAC and ground.	28 volts dc.	Replace airstream direction detector. (Refer to paragraph 6-211.)
		Zero volts.	Continue trouble shooting.
No d-c power to heater element.	Check between test points FAE and FAF and ground and between test points GL and GK. **Note** Test point GL is positive.	28 volts dc.	Replace defective GROUND SAFETY RELAY NO. 1.
		Zero volts between test point FAE and ground and between test point FAF and ground.	Replace defective d-c power wire(s).
		Zero volts between test points GL and GK.	Refer to paragraph 3-102, Trouble Shooting Landing Gear System.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF YAW HEATER and GROUND SAFETY* (GROUND SAFETY & OVERVOLTAGE†) circuit breakers engaged.
Airplane on jacks or ground safety switch linkage disconnected and moved to the unloaded position.

CAUTION

Do not allow angle-of-sideslip probe to overheat.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
ANGLE-OF-SIDESLIP PROBE HEATER INOPERATIVE.			
Heater element defective.	Check between test point FAD and ground.	28 volts dc.	Replace airstream direction detector. (Refer to paragraph 6-211.)
		Zero volts.	Continue trouble shooting.
No d-c power to heater element.	Check between test points FAE and FAF and ground and between test points GL and GK. **Note** Test point GL is positive.	28 volts dc.	Replace defective GROUND SAFETY RELAY NO. 1.
		Zero volts between test point FAE and ground and between test point FAF and ground.	Replace defective d-c power wire(s).
		Zero volts between test points GL and GK.	Refer to paragraph 3-102, Trouble Shooting Landing Gear System.

*Airplanes 139531i through 139555i
†Airplanes 141444j and subsequent

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND, ANGLE OF ATTACK HEATER, ANGLE OF YAW HEATER, GROUND SAFETY* (GROUND SAFETY & OVERVOLTAGE†) and NO. 1 INV PWR circuit breakers engaged.
INST. AC POWER switch in "NO. 1 INV." position.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
D-C AND/OR A-C POWER FAILURE.			
Circuit-breaker failure.	Check between test points PGT, PGU, PGW, PGR and ground.	28 volts dc.	Replace defective circuit breaker(s).
		Zero volts.	Refer to paragraph 8-61, Trouble Shooting D-C Power Distribution System.
Fuse failure.	Check between test point XAB and ground.	115 volts ac.	Replace defective fuse.
		Zero volts.	Refer to paragraph 8-78, Trouble Shooting A-C Power Supply and Distribution System.

*Airplanes 139531i through 139555i
†Airplanes 141444j and subsequent

6-202. OPERATIONAL CHECK OF ANGLE-OF-ATTACK INDICATING SYSTEM.

6-203. To check the operation of the angle-of-attack indicating system, proceed as follows:

Note

- Three men are necessary to perform an operational check.
- Steps a. through e. are performed with power off.

a. Remove protective covers from the probes of the angle-of-attack and the angle-of-sideslip airstream direction detectors.

b. Examine the probes; there should be no dents or other evidence of damage.

Note

If a probe is covered with ice, follow instructions in steps e. through k. before proceeding with the check. Allow the probe heaters to be energized only long enough to loosen ice so that it can be removed by hand; then, repeat procedure step b.

c. With the aid of a short length of tubing, blow directly into each slot of probe to check freedom of probe rotation. (Do not blow into both slots at once.) The probe should rotate in the air stream.

d. Rotate each probe by hand from one limit stop to the other. There should be no evidence of roughness, sticking or binding. The tip of the probe should not describe an arc when it is viewed endwise and rotated.

e. Place the D.C. POWER switch, located on the right-hand forward console, in the "OFF" position.

f. Connect a source of external power to the airplane.

g. Depress the ANGLE OF ATTACK IND, ANGLE OF ATTACK HEATER, ANGLE OF YAW HEATER and GROUND SAFETY [on airplanes 139531i through 139555i (GROUND SAFETY & OVERVOLTAGE on airplanes 141444j and subsequent)] circuit breakers located on the left-hand radio bay circuit-breaker panel.

h. Feel the case of the angle-of-attack airstream direction detector (remove left-hand gun bay door to feel the unit). The case should be warm to the touch.

Note

The internal heater in the angle-of-attack airstream direction detector should be energized whenever the ANGLE OF ATTACK IND circuit breaker is depressed and there is 28-volt d-c power to the system. Thermal cutout for the internal heater is 40.5°C (105°F).

i. Feel the probes of the airstream direction detectors; they should *not* be warm to the touch.

j. Disable the ground safety switch on the left-hand landing gear to energize the ground safety relays. To disable the switch, remove cotter pin, washer and pin securing actuator link to switch arm. Install warning flag on switch arm. (Refer to General Information, Section VI.)

WARNING

Make certain the STORES JETTISON & DROP TANK TRANSFER circuit breaker, located on the left-hand forward console, is pulled.

Contacts on ground safety relay No. 1 will complete the circuits to the two probe heaters and to the internal heater of the angle-of-sideslip detector. After 30 seconds, the probes should be warm to the touch.

CAUTION

Severe burns may be received, or probes damaged, if probe heaters are energized for long periods of time. Thermal cutout for the probe heaters is 93°C (200°F).

k. Reverse procedure for disabling ground safety switch (step j.).

l. Examine the angle-of-attack indicator and check operation of the movable index (or "bug") over the dial by rotating the set knob located at the lower left-hand side on the front of the indicator.

m. Manually rotate the angle-of-attack probe from one limit stop to the other. Make certain that the indicator pointer follows probe rotation. Facing the airplane, rotate the probe clockwise; the indicating pointer should also move clockwise or toward the "—5" end of the scale. Rotate the probe counterclockwise to the limit stop; the pointer should move to the "30" end of the scale.

n. Rotate the probe from one limit stop to the other at a rate of approximately 3 seconds. The indicator pointer should sweep the scale in approximately 5 seconds.

o. Depress the EXTERIOR LIGHTS circuit breaker located on the right-hand forward console panel.

oA. On airplanes 143594m and subsequent, the approach indexer will illuminate at this point. Rotate the angle-of-attack probe from one limit to the other, observing the approach indexer and the angle-of-attack indicator. The approach indexer lamps should illuminate with the corresponding angle-of-attack indicator readings as follows:

APPROACH INDEXER LIGHT	ANGLE-OF-ATTACK INDICATOR READING
Top	19.5 to 30 units
Top and center	19 to 19.5 units
Center	17 to 19 units
Center and bottom	16.5 to 17 units
Bottom	5 to 16.5 units

oB. Depress the ARREST HOOK circuit breaker located on the right-hand forward console panel.

p. Place the arresting hook control handle, located forward of the right-hand console panel, in the "UP" position.

q. Place the RUD. PED. SHAKER TEST AND APP LIGHT HOOK BY-PASS switch, located on the test switch panel assembly on the right-hand canopy deck, in the "APP LIGHT HOOK BY-PASS" position.

r. Position the EXTERIOR LIGHTS switch, located forward of the power control lever on the left-hand forward vertical console, to "OFF."

s. Rotate the angle-of-attack probe clockwise from its lower limit to its upper limit observing the approach lights installed in the nose of the airplane. The lights should change from green through amber to red. Rotate the probe counterclockwise from its upper limit to its lower limit. The lights should change from red through amber to green. Check that the lights change at the correct unit indication on the angle-of-attack indicator at change-over points as follows:

DIAL INDICATION	APPROACH LIGHT
—5.0 to 17.0 (±0.3)	RED
17.0 (±0.3) to 19.0 (±0.3)	AMBER
19.0 (±0.3) to 30	GREEN

t. Position the EXTERIOR LIGHTS switch to the "ON" position to energize the approach light dimming relay. Repeat procedure step s. to check the approach light dimmed circuits.

u. Depress the WARNING LIGHT TEST & RUD. PED. SHAKER circuit breaker located on the right-hand forward console panel.

v. Position the RUD. PED. SHAKER TEST AND APP LIGHT HOOK BY-PASS switch to the "RUD. PED. SHAKER TEST" position and *hold*.

w. Rotate the angle-of-attack probe counterclockwise from its upper limit stop to its lower limit stop. Observe the angle-of-attack indicator and the rudder pedal shaker on the right rudder pedal; at 19.0 (±0.3) units dial indication, the rudder pedal shaker should be actuated and should continue shaking to 30 units dial indication.

x. Rotate the probe clockwise and check that the rudder pedal shaker stops shaking at 19.0 (±0.3) units dial indication.

y. Remove the source of external power.

z. Replace the protective covers on the probes of the angle-of-attack and angle-of-sideslip airstream direction detectors.

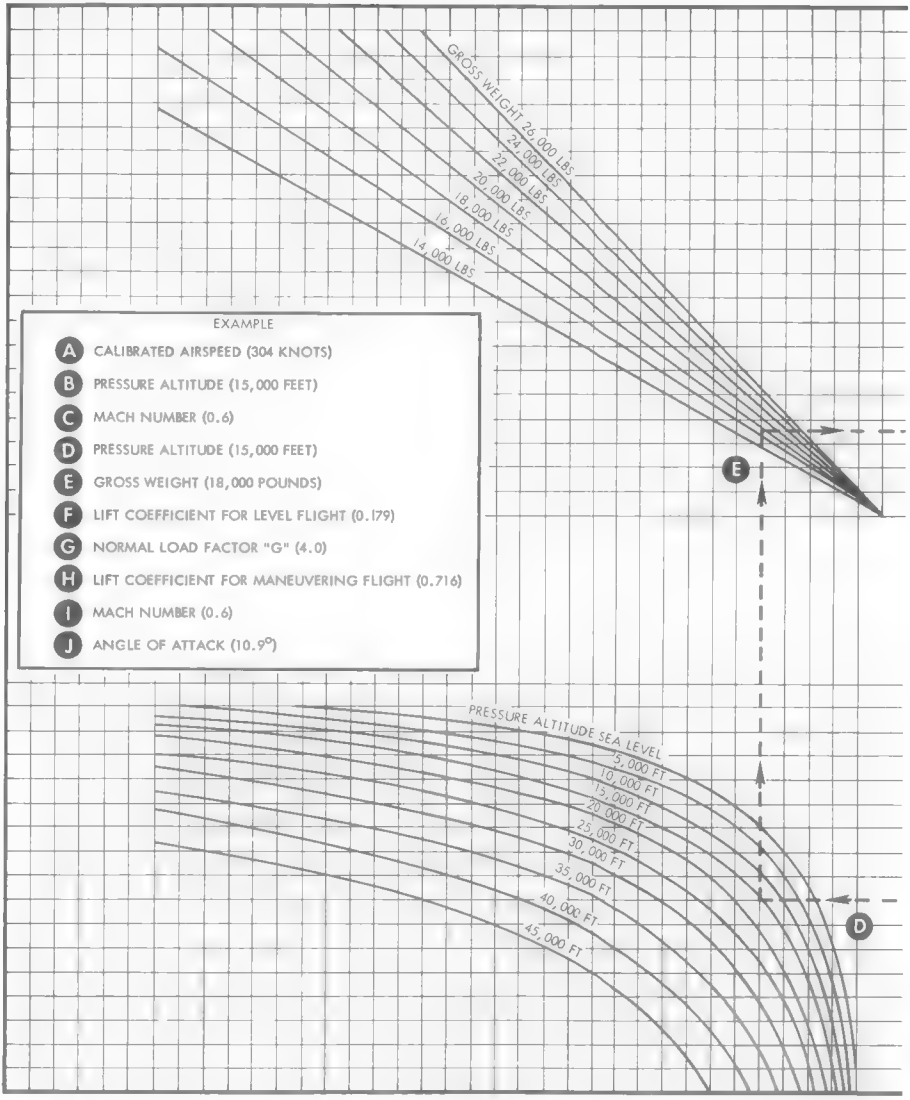


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Figure No. 6-43. Angle-of-Attack Relationship Curves (Sheet 1)

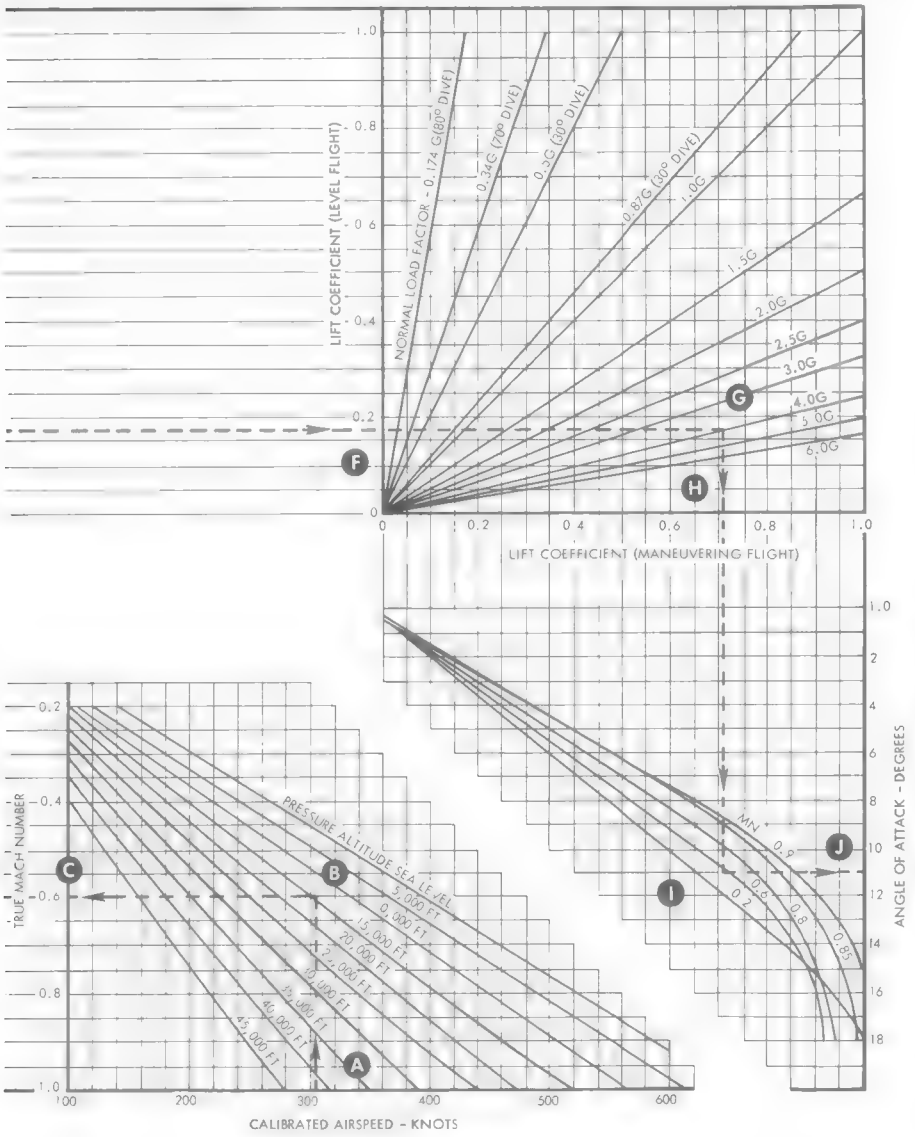

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Figure No. 6-43. Angle-of-Attack Relationship Curves (Sheet 2)

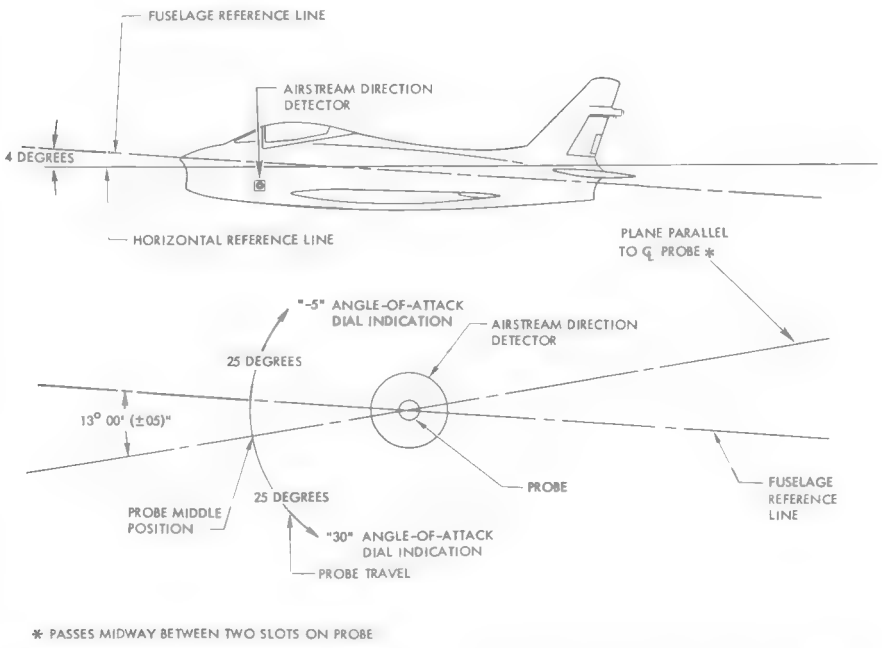


Figure No. 6-44. Airstream Direction Detector Installation Relative to the Fuselage Reference Plane

6-204. ANGLE-OF-ATTACK INDICATIONS AND AIRPLANE REFERENCE LINES.

6-205. Since the angle-of-attack indication is derived from the angular difference between local airflow and the fuselage reference plane, the dial unit indication is only relative to the true angle of attack of the airplane. The airstream direction detector, located on the left side of the fuselage, is installed so that a plane parallel to the centerline of the probe and midway between the two slots is inclined 13 degrees (±5 minutes) below the fuselage reference plane. The relationship of the installation of the airstream direction detector to the fuselage reference plane is shown in figure 6-44.

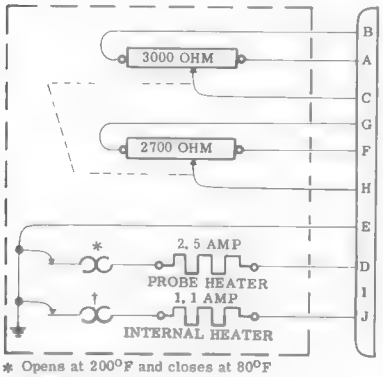
6-206. ANGLE-OF-ATTACK RELATIONSHIP CURVES.

6-207. The relative changes in angle of attack, as a result of changes in power settings and changes in calibrated airspeed (1 G flight), may be seen by examining the angle-of-attack relationship curve chart. (See figure 6-43.) From the illustrated example, it should be noted that the higher angles of attack (in mils) vary inversely with changes in power settings. Variations in angle of attack with drop tanks as compared to a clean airplane, or with speed brakes open as compared to speed brakes closed, are negligible and can, therefore, be ignored during interpretation of chart information.

6-208. ANGLE-OF-ATTACK AIRSTREAM DIRECTION DETECTOR.

6-209. The airstream direction detector, which is installed on the left side of the forward fuselage, will measure direction of airflow within an accuracy of 0.1 degree with an operating range of 90 knots to Mach 3.0. The body of the detector is mounted within the fuselage with the probe extending outward through the skin of the airplane. (See figure 6-41.) The probe is free to rotate through an angle of 50 degrees. To ensure the proper angular positioning of the probe and, subsequently, the correct indication of relative angle of attack, the mounting flange of the detector contains two dowel pins. These pins must be fitted to the mating holes on the mounting plate. Filters are installed at the

base of the probe to keep dirt from the air passages out of the paddle chamber. A 60-watt heater element, which extends the length of the separator in the probe, is used for de-icing the unit. This heater circuit is routed through ground safety relay No. 1 (controlled by the ground safety switch) to protect the element when the airplane is on the ground. The ANGLE OF ATTACK HEATER circuit breaker, located on the left-hand radio bay circuit-breaker panel, should be pulled at all times unless the heater is to be used. A thermostatically controlled internal heater eliminates condensation in the body of the detector unit. This heater is energized whenever the system is in operation, but the thermostat opens the circuit at 40.5°C (105°F).



* Opens at 200°F and closes at 80°F
† Opens at 105°F

FJ-48-2-51-196

Figure No. 6-44A. Angle-of-Attack Airstream Direction Detector—Internal Schematic

6-210. CHECKING ANGLE-OF-ATTACK AIRSTREAM DIRECTION DETECTOR OPERATION. When a new airstream direction detector is installed, it is necessary to perform an operational check of the angle-of-attack indicating system (paragraph 6-202) to make sure the detector is aligned with the indicator. If the same detector is reinstalled, test the operation of the unit to see that it has been installed correctly. To check the detector, proceed as follows:

a. Place the D.C. POWER switch, located on the right-hand forward console panel, in the "OFF" position.

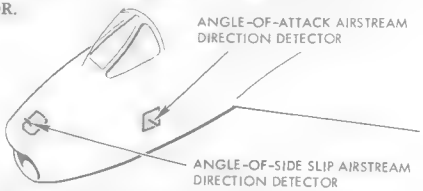
b. Connect a source of external power to the airplane.

c. Check that the ANGLE OF ATTACK IND circuit breaker, located on the left-hand radio bay circuit-breaker panel, is depressed.

d. Manually rotate the probe from one limit stop to the other. There should be no evidence of roughness, sticking or binding.

e. The indicator should follow probe rotation.

6-211. REMOVING AND INSTALLING ANGLE-OF-ATTACK AIRSTREAM DIRECTION DETECTOR.



Note This procedure also applies to the angle-of-side slip airstream direction detector.

REMOVING

Caution Before removing the airstream direction detector, make certain all power to the instruments is off.

1. Remove the protective cover from the probe by carefully pulling it straight out from the fuselage.

Caution Exercise care to avoid bending the probe when removing the protective cover.

2. Remove screws from fuselage cover plate and remove cover plate.

3. Remove four bolts from airstream direction detector retaining plate and carefully remove plate.

Caution Exercise care that the detector unit is not pulled loose and dropped when the retaining plate is removed.

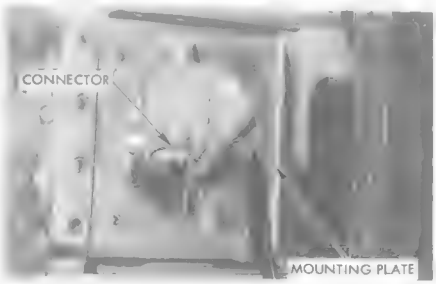
4. Remove airstream direction detector by pulling it free of cutout in mounting panel.

5. Remove electrical connector from electrical bracket located adjacent to detector mounting panel. Cover plug and receptacle with masking tape.

6. Remove detector from the airplane.

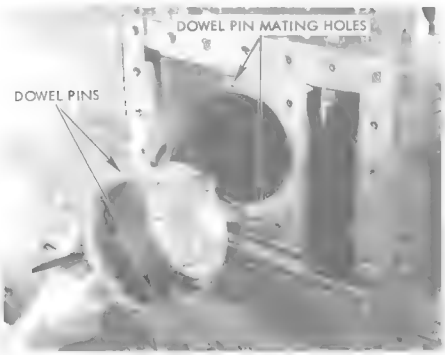
INSTALLING

1. Remove masking tape from electrical plug and receptacle and make electrical connection. Safety-wire connection with AN995F32 lockwire.

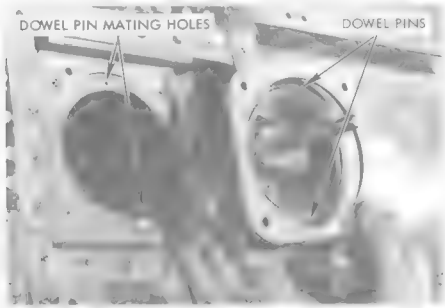


2. Position detector in cutout in mounting plate making certain the dowel pins on the detector are inserted in the mating holes on the mounting plate.

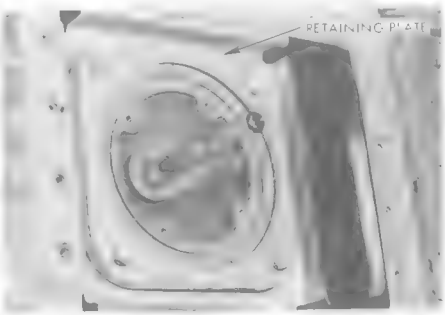
FJ-48-2-51-135



Note Two dowel pins on the detector must be inserted in the mating holes on the mounting plate for correct angular position and operation of the airstream direction detector.



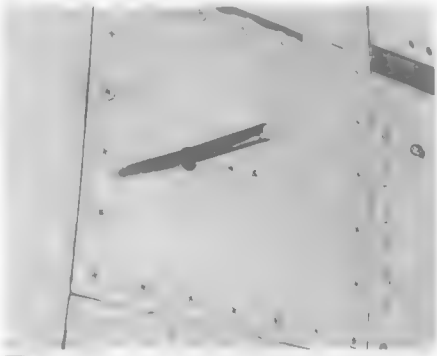
3 As soon as the detector is positioned, install the retaining plate and secure with four mounting bolts.



FJ-4B-2-51-136

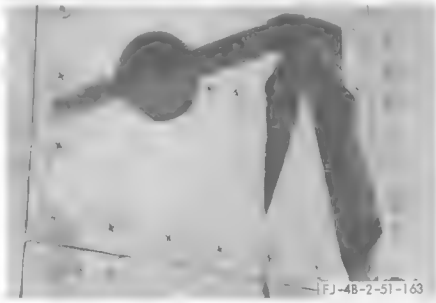
4 Carefully position the fuselage cover plate over the probe of the detector unit.

5 Install and tighten the screws in the cover plate.



6 If a new detector was installed, perform an operational check of the system. (Refer to paragraph 6-202.) If the same detector was reinstalled, perform an operational check of the detector. (Refer to paragraph 6-210.)

7 Install protective cover over probe.

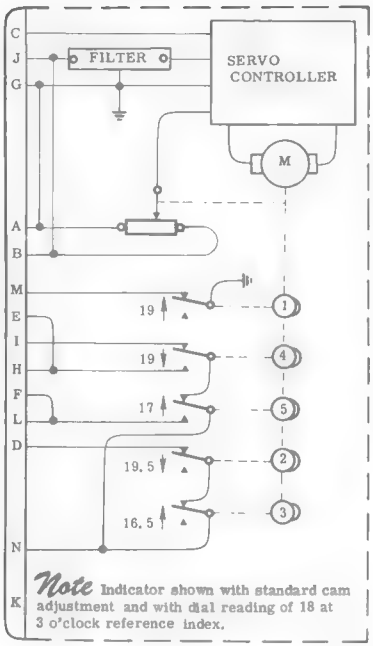


6-212. ANGLE-OF-ATTACK INDICATOR.

6-213. The angle-of-attack indicator, located in the upper left-hand corner of the instrument panel, indicates the relative angle of attack of the airplane as measured at the angle-of-attack detector. An indicating pointer moves over a scale which is graduated from −5 to +30. A movable index mark or "bug" can be positioned on the scale at specific points to indicate the optimum approach angle of attack or it can be used for guiding or warning purposes. The "bug" is set by means of an external knob located at the lower left-hand corner of the front of the instrument. A fixed reference index is located on the face of the indicator at the three o'clock position. Whenever the indicator pointer is within the limits of the index (limits are 17 to 19 units), an optimum angle of attack is indicated. The indicator houses a servo controller, a servomotor, a receiver potentiometer, five adjustable cams and five switches. The switches are actuated by the cams at preset angles of attack (paragraphs 6-231 and 6-238) and a circuit is completed to the appropriate approach light. If the indicated angle of attack is between 19 and 30 units, the circuit to the rudder pedal shaker is also completed. On airplanes 143594m and subsequent, circuits are also available to illuminate the bulbs of the approach indexer.

6-214. COMPENSATED ANGLE-OF-ATTACK AND ANGLE-OF-SIDESLIP SYSTEM.

6-215. The compensated angle-of-attack and angle-of-sideslip system is composed of an angle-of-sideslip airstream direction detector, located forward of the nose equipment compartment above the air intake duct, and an angle-of-attack and angle-of-sideslip compensator, located in the nose equipment compartment. The system also utilizes the angle-of-attack airstream direction detector and the pressure ratio transducer. The purpose of the system is to convert local angle-of-attack and local angle-of-sideslip signals into signals which correspond with true angle of attack and true angle of sideslip. These signals which represent true angle of attack and true angle of sideslip are then conveyed to the fire control system for controlled rocket firing. Power for the system is 28 volts dc from the primary bus and 115 volts, 400-cycle, phase "A" from the primary a-c bus. The d-c circuit is protected by a 5-ampere



FJ-4B-2-51-195

Figure No. 6-44B. Angle-of-Attack Indicator—Internal Schematic

circuit breaker (ANGLE OF ATTACK IND), located on the left-hand radio bay circuit-breaker panel and the a-c circuit is protected by a one-ampere fuse located on the right-hand rear vertical console panel. (See figure 6-42.)

CAUTION

Do not use the angle-of-sideslip airstream direction detector as a step or handgrip. This could easily result in bending the probe, causing it to stick or bind and rendering the system inoperable or inaccurate.

Note

A cover is provided for the probe of the angle-of-sideslip airstream direction detector. This cover should be installed at all times when the airplane is on the ground to keep moisture and dirt from entering the probe.

CAUTION

Remove probe cover from the angle-of-sideslip airstream direction detector prior to engine run.

6-216. ANGLE-OF-SIDESLIP AIRSTREAM DIRECTION DETECTOR.

6-217. The angle-of-sideslip airstream direction detector is mounted forward of the nose equipment compartment above the air intake duct. The center of rotation coincides with the longitudinal axis of the airplane. The angle-of-sideslip airstream direction detector is similar to the angle-of-attack airstream direction detector (paragraph 6-208) except that it has only one potentiometer. Signals from this potentiometer along with signals from one potentiometer in the angle-of-attack airstream direction detector are supplied to the angle-of-attack and angle-of-sideslip compensator to furnish the compensator with local angle-of-attack and local angle-of-sideslip information. No visual indication of the angle of sideslip is presented to the pilot. For information concerning removal and installation of the angle-of-sideslip airstream direction detector, refer to paragraph 6-211.

6-218. ANGLE-OF-ATTACK AND ANGLE-OF-SIDESLIP COMPENSATOR.

6-219. The angle-of-attack and angle-of-sideslip compensator, located on the right-hand side of the radar equipment shelf in the nose equipment compartment, converts local angle-of-attack and local angle-of-sideslip signals into true angle-of-attack and true angle-of-sideslip signals. Local angle-of-attack and local angle-of-sideslip signals are received from the airstream direction detectors. Since the relationship between local angle of attack and local angle of sideslip and the corresponding true angle of attack and true angle of sideslip varies with Mach number, corrections are made for Mach number by three resistance ratio inputs from the pressure ratio transducer. The compensator also receives a reference voltage from the Mark 86 Mod 0 computer and modifies the voltage according to the true angle of attack and true angle of sideslip. Two voltages, proportional to true angle of attack and true angle of sideslip, are then sent to the computer. These voltages alter the position of the gyro pipper in the sight unit to compensate for the airplane's angle of attack and angle of sideslip. Two dials, calibrated in mils, are located on the face of the compensator. These are bore-sighting adjustments used to correct for minor airframe variations.

6-220. BORE SIGHTING COMPENSATED ANGLE-OF-ATTACK AND ANGLE-OF-SIDESLIP SYSTEM.

6-221. Bore sighting the angle-of-attack and angle-of-sideslip system should be accomplished whenever any component or components in the system have been replaced. Bore sighting may be accomplished after the airplane has been flown and an in-flight angle-of-attack

check performed. [Refer to the FJ-4 and FJ-4B Supplemental Flight Handbook (NAVAER 01-60JKD-501A).] Two dials, calibrated in mils, are located on the face of the compensator for this purpose. If the gyro pipper is observed to be too high, the ATTACK dial is rotated counterclockwise the number of mils required and the pipper is moved downward. If the pipper is reported to be too low, the ATTACK dial is moved clockwise the number of mils required, thus moving the pipper up. Should an error be observed in azimuth, the SIDESLIP dial is used. However, only 85 percent of the correction required is set into the compensator by the SIDESLIP dial. If the pipper is found to be too far to the right, the SIDESLIP dial is turned counterclockwise 85 percent of the correction required, and the pipper moves to the left. If the pipper is observed to be too far to the left, the SIDESLIP dial is rotated clockwise 85 percent of the correction required, moving the pipper to the right.

6-222. FUNCTIONAL TEST OF ANGLE-OF-ATTACK AND ANGLE-OF-SIDESLIP COMPENSATOR AND TIE-IN TESTING OF COMPENSATED ANGLE-OF-ATTACK AND ANGLE-OF-SIDESLIP SYSTEM WITH MARK 16 FIRE CONTROL SYSTEM.

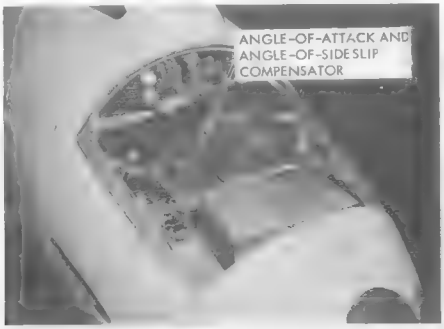
6-223. Information will be supplied when available.

6-224. REMOVING AND INSTALLING ANGLE-OF-ATTACK AND ANGLE-OF-SIDESLIP COMPENSATOR.

REMOVING

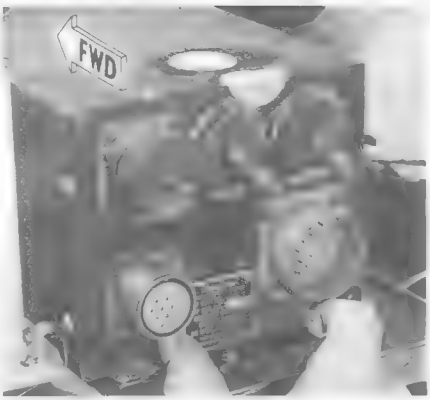
1 Remove nose equipment compartment access door.

2 Cut safety wire and disengage two electrical connectors from the angle-of-attack and angle-of-sideslip compensator located on the right side of the radar equipment mounting shelf.



3 Cut safety wire and disengage two snap slide fasteners from studs on aft end of compensator and pull the compensator aft to free the pin assemblies.

FJ-48-2-51-138



4 Remove the compensator from the airplane.

INSTALLING

1 Place the compensator on the mounting tray on the right side of the radar equipment mounting shelf and position against the two pin assemblies.

2 Engage snap slide fastener with stud on each aft corner of the compensator and safety with AN995F32 lockwire.

3 Make two electrical connections and safety connectors with AN995F32 lockwire.

4 Replace access door.

FJ-48-2-51-137

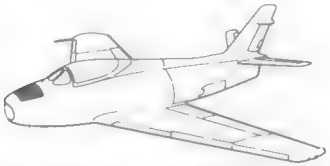
6-225. PRESSURE RATIO TRANSDUCER.

6-226. The pressure ratio transducer, installed in the nose equipment compartment, provides functions of Mach to the angle-of-attack and angle-of-sideslip compensator and to the Mach sensing trim system. (See figure 6-42.) The transducer consists of a bellows and beam displacement sensor arrangement, a d-c motor and four cam-driven potentiometers. (See figure 6-45.) Pitot and static pressures from the pitot-static system are admitted to the bellows and as changes in pressure in the bellows occur, the beam is rotated to trigger the sensor switch which, in turn, operates the motor. The motor drives the traveling fulcrum along the beam until a moment balance exists which releases contact pressure at the switch, de-energizing the motor. When the fulcrum is repositioned by the motor, mechanical linkage rotates the cams which, in turn, drive the potentiometers to the correct value. Three of the cam-driven potentiometers furnish Mach number functions to the angle-of-attack and angle-of-sideslip compensator; one cam-driven potentiometer is an integral part of the Mach sensing trim system. (Refer to paragraph 2-284.)

6-227. REMOVING AND INSTALLING PRESSURE RATIO TRANSDUCER.

REMOVING

1 Remove nose compartment access door.



2 Remove electrical connectors from the KY81/APA-89 coder unit and the angle-of-attack and angle-of slide slip compensator, located on the radar equipment mounting shelf.

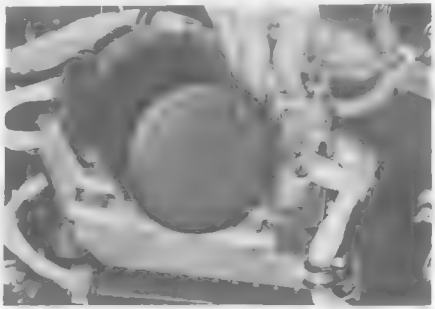
3 Pull two pins from the left-hand side of the radar equipment mounting shelf.

FJ-48-2-51-117

4 Carefully swing radar equipment shelf into the upright position.

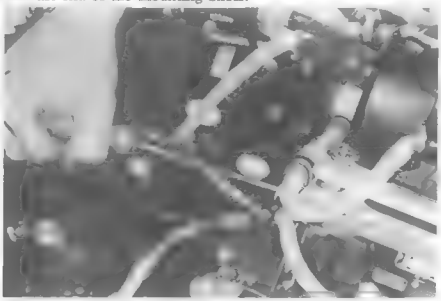
5 Remove pitot and static pressure lines from the 90-degree fittings aft and inboard of the pressure ratio transducer.

6 Disengage snap slide fasteners on each corner of the pressure ratio transducer mounting shelf; rotate forward end of the mounting shelf inboard and remove the two electrical connectors.



FJ-48-2-51-118

7 Remove two wire bundle securing clamps from the aft end of the mounting shelf.



8 Remove the inboard mounting screw from each side of the longitudinal trim amplifier, loosen the two outboard screws and allow the unit to rotate downward.

9 Remove four pressure ratio transducer mounting screws from the underside of the mounting shelf and lift the pressure ratio transducer from the airplane.

INSTALLING

1 Position pressure ratio transducer on mounting shelf and install four mounting screws.

2 Reposition the longitudinal trim amplifier and install and tighten mounting screws.

3 Install two electrical connectors and safety with AN995F32 lockwire.

4 Position mounting shelf, secure snap slide fasteners on each corner of the mounting shelf and safety with AN995F32 lockwire.

5 Secure wire bundle on aft end of mounting shelf with two clamps.

6 Connect pitot and static pressure lines.

7 Gently lower radar equipment mounting shelf and install two pins on left-hand side.

8 Install electrical connectors on KY81 APA-89 coder unit and angle-of-attack and angle-of-side slip compensator. Safety with AN995F32 lockwire.

9 Install nose compartment access door.

FJ-48-2-51-119

6-228. RUDDER PEDAL SHAKER SYSTEM.

6-229. The rudder pedal shaker system is provided as a stall warning to the pilot. The system consists of a vibrator-motor installed on the right rudder pedal and actuated by a cam in the angle-of-attack indicator. The cam is set to energize the vibrator-motor at a predetermined and preset angle of attack. The circuit to the rudder pedal shaker is routed through ground safety relay No. 1 which prohibits operation of the vibrator-motor when the airplane is on the landing gear. A test switch (RUD. PED. SHAKER TEST AND APP LIGHT HOOK BY-PASS), located on the test switch panel assembly on the right-hand canopy deck, is provided to check the system. Operating power for the system is 28 volts dc from the primary bus. The circuit is protected by a 5-ampere circuit breaker (WARNING LIGHT TEST & RUD. PED. SHAKER) located on the right-hand console panel.

6-230. TROUBLE SHOOTING RUDDER PEDAL SHAKER SYSTEM.

TEST EQUIPMENT: D-C voltmeter.
Ohmmeter.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
WARNING LIGHT TEST & RUD. PED. SHAKER and GROUND SAFETY* (GROUND SAFETY & OVERVOLTAGE†) circuit breakers engaged.
Airplane on jacks or ground safety switch linkage disconnected and moved to the unloaded position.

Note

Desired meter readings will be available only when angle-of-attack airstream direction detector is positioned to provide a reading of 19 to 30 units on the angle-of-attack indicator.

*Airplanes 139531i through 139555i
†Airplanes 141444j and subsequent

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
RUDDER PEDAL SHAKER INOPERATIVE.			
Defective rudder pedal shaker.	Check between test points CPA and CPB. **Note** Test point CPB is positive.	28 volts dc.	Replace rudder pedal shaker. (Refer to paragraph 6-234.)
		Zero volts.	Continue trouble shooting.
Angle-of-attack indicator defective or out of adjustment.	Check between test point CPA and ground.	Zero ohms.	Continue trouble shooting.
		Other than zero ohms.	Perform wire segment continuity check; if satisfactory, readjust or replace angle-of-attack indicator. (Refer to paragraphs 6-7 and 6-8.)
Defective GROUND SAFETY RELAY NO. 1 or defective associated wiring.	Check between test point CPC and ground.	28 volts dc.	Continue trouble shooting.
		Zero volts.	Replace defective power wire.
	Check between test points GL and GK. **Note** Test point GL is positive.	28 volts dc.	Replace defective GROUND SAFETY RELAY NO. 1.
		Zero volts.	Continue trouble shooting.
	Check between test point GL and ground.	28 volts dc.	Continue trouble shooting.
		Zero volts.	Replace defective power wire.
	Check between test point GDE and ground.	Zero ohms.	Repair or replace defective wire to test point GK.
		Other than zero ohms.	Refer to paragraph 3-102, Trouble Shooting Landing Gear System.
RUDDER PEDAL SHAKER OPERATES ON THE GROUND WITH ANGLE-OF-ATTACK SYSTEM INOPERATIVE.			
Defective RUD. PED. SHAKER TEST AND APP LIGHT HOOK BY-PASS switch.		None.	Replace defective RUD. PED. SHAKER TEST AND APP LIGHT HOOK BY-PASS switch.
D-C POWER FAILURE.			
Circuit-breaker failure.	Check between test points PGR and PBK and ground.	28 volts dc.	Replace circuit breaker(s).
		Zero volts.	Refer to paragraph 8-61, Trouble Shooting D-C Power Distribution System.

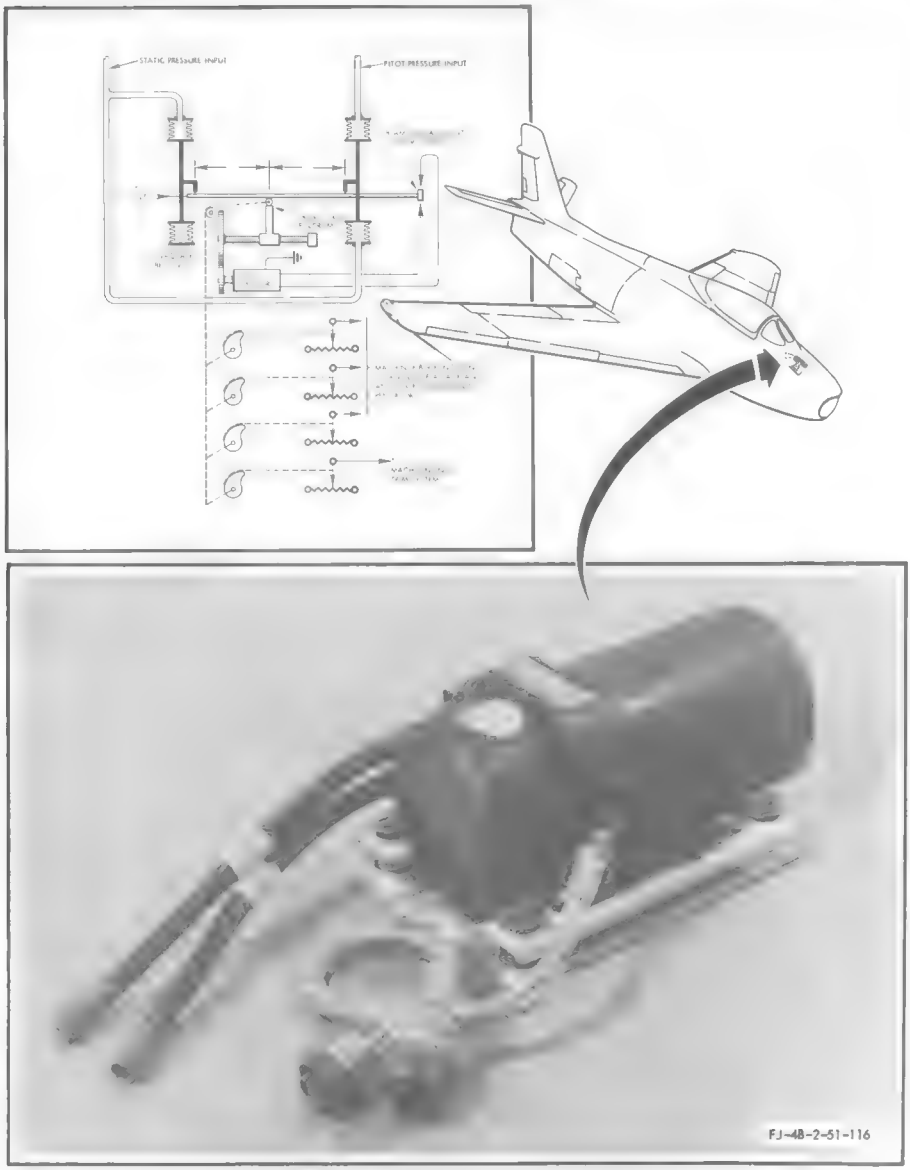


Figure No. 6-45. Pressure Ratio Transducer

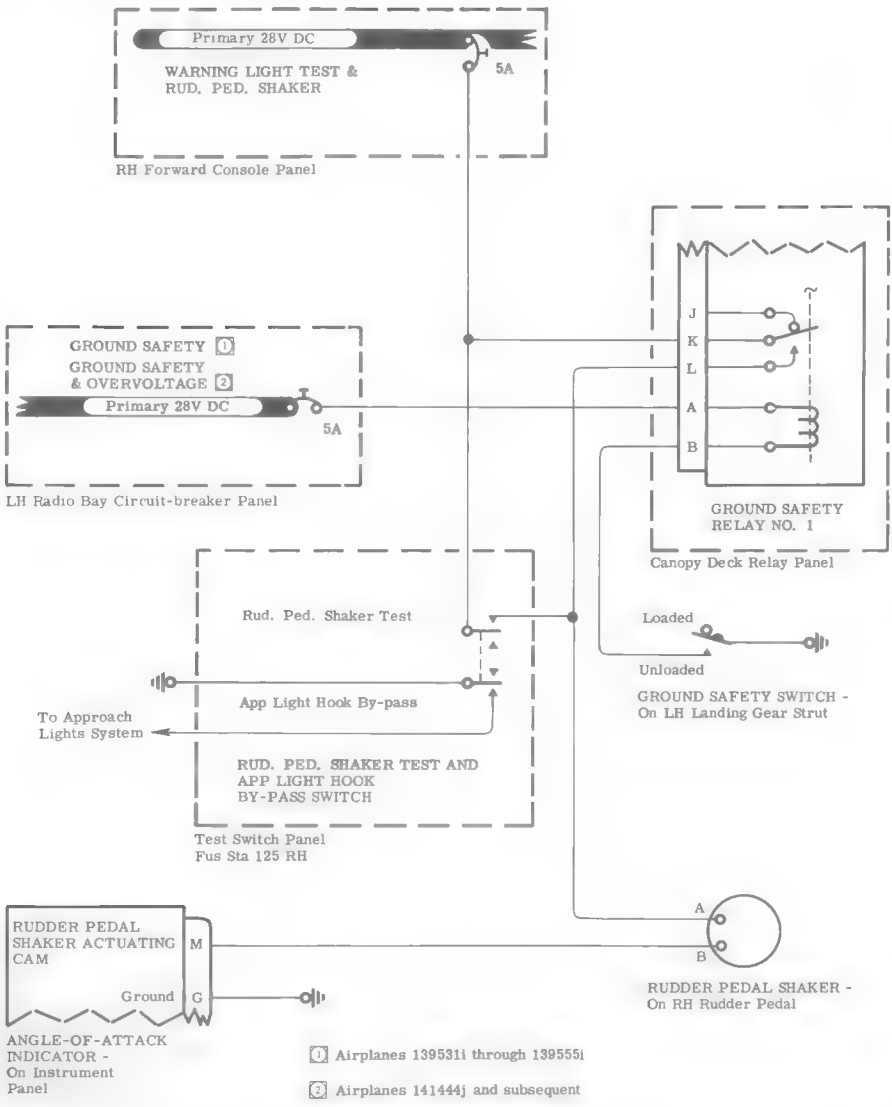
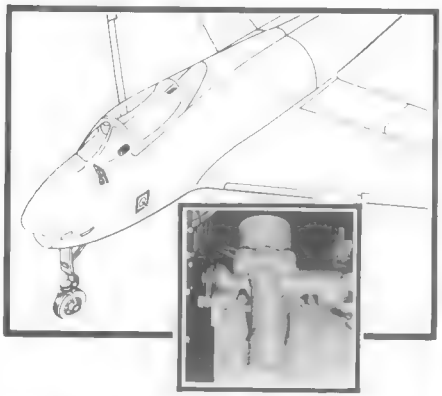


Figure No. 6-46. Rudder Pedal Shaker System Schematic

FJ-4B-2-51-185A

6-231. ADJUSTING RUDDER PEDAL SHAKER SYSTEM.



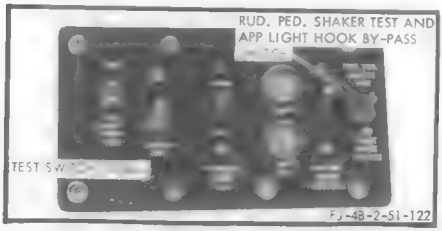
Note Two men are required to check and adjust the rudder pedal shaker system.

1 Place the d-c power switch, located on the right-hand forward console, in the "OFF" position.

2 Engage the ANGLE-OF-ATTACK circuit breaker located on the left-hand radio bay circuit-breaker panel. Engage the WARNING LIGHT TEST & RUD. PED. SHAKER circuit breaker located on the right-hand forward console.

3 Connect a source of external power to the airplane.

4 Place the RUD. PED. SHAKER TEST AND APP LIGHT HOOK BY-PASS switch, located on the test switch panel, in the "RUD. PED. SHAKER TEST" position and hold.



5 Remove protective cover from airstream direction detector probe. From extreme clockwise position, slowly rotate probe counterclockwise. The indicator pointer will move from -5 units toward 30 units. When pointer reaches 19.0 (±0.3) rudder pedal shaker should become operative and remain operative until pointer drops below 19.0 units again.

6 If actuation point is 19.0 (±0.3) on dial, setting of the accessory cam adjustment in angle-of-attack indicator is correct; if not, proceed to step 7.

7 Disengage ANGLE OF ATTACK IND and WARNING LIGHT TEST & RUD. PED. SHAKER circuit breakers.

8 Remove angle-of-attack indicator from instrument panel. (Refer to paragraph 6-7.)

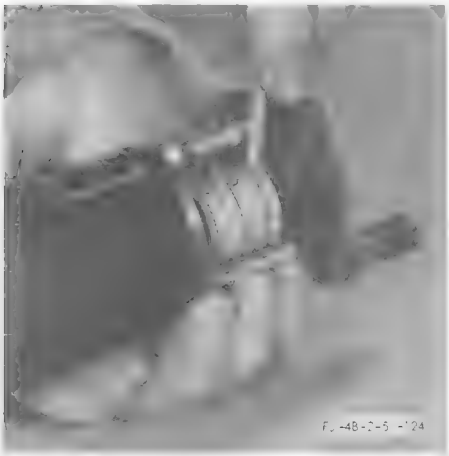
9 Cut safety wire and remove three cover securing screws from rear of indicator.

10 Remove cylindrical cover to expose cam adjustment.

Note

- Cam No. 1 (nearest face of indicator) is the adjustment cam for the rudder pedal shaker system.

- Changing switching point of cam is accomplished by inserting 1/8-inch dowel pin or drill rod in radial hole in side of cam and using it as a lever to rotate cam relative to camshaft.



11 Reconnect the power source to the rear of the angle-of-attack indicator.

12 Engage ANGLE OF ATTACK IND and WARNING LIGHT TEST & RUD. PED. SHAKER circuit breakers.

13 Rotate probe until indicator reads 19.0 (±0.3) units.

14 Adjust cam until the rudder pedal shaker becomes energized.

15 Repeat steps 5 and 6.

16 Remove external power source.

17 Remove power source from rear of angle-of-attack indicator.

18 Replace cover on angle-of-attack indicator and install three cover securing screws. Safety-wire screws with AN995F32 wire.

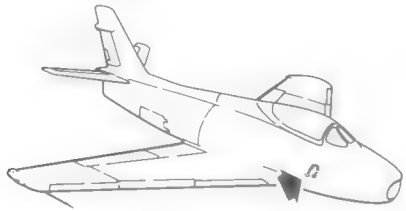
19 Install angle-of-attack indicator. (Refer to paragraph 6-8.)

FJ-48-2-51-125

6-232. RUDDER PEDAL SHAKER.

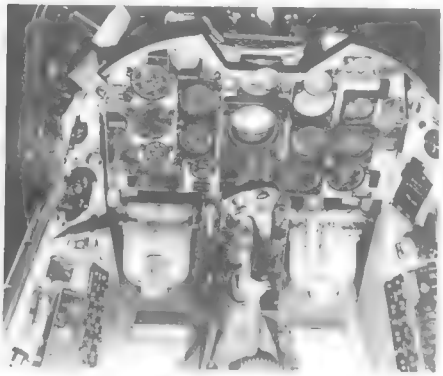
6-233. The rudder pedal shaker is a conventional motor which incorporates an eccentrically weighted rotating shaft that produces a vibrating force. The operating speed is 1150 to 1300 rpm. The motor will vibrate with a shaking force of 3.25 to 4.75 pounds.

6-234. REMOVING AND INSTALLING RUDDER PEDAL SHAKER.



REMOVING

1. Place rudder pedals in gust locks. (Refer to paragraph 2-190.)



2. Remove two nuts securing clamp on forward end of rudder pedal shaker and remove clamp.
3. Remove two nuts securing strap over aft end of rudder pedal shaker and remove strap.
4. Put instrument panel in servicing position by rotating the stud fasteners on front of instrument panel counterclockwise and allowing panel to come aft to position where safety cords are taut.
5. Follow electrical leads from rudder pedal shaker and remove clamps securing wire bundle.

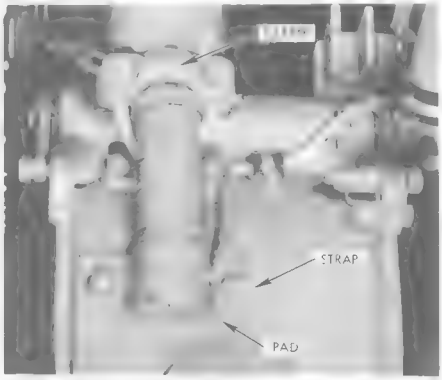
Caution Make sure all power is off.

6. Remove leads from terminal strip No. 39.
7. Carefully remove rudder pedal shaker and wiring from airplane.

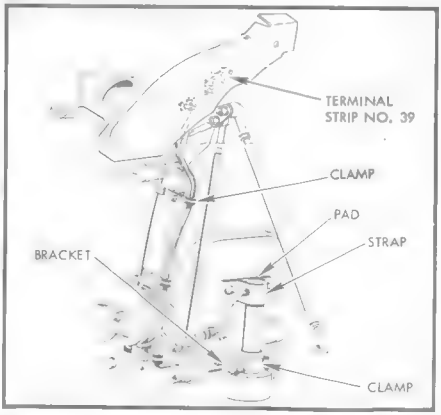
FJ-48-2-51-120

INSTALLING

1. Position rudder pedal shaker in bracket with aft end resting on pad.
2. Position clamp on forward end and secure with two nuts.
3. Position strap over aft end and secure with two nuts.



4. Install electrical wiring as shown and connect leads to terminal strip No. 39.



5. Replace instrument panel to mounted position and rotate stud fasteners clockwise.
6. Remove rudder pedals from gust locks. (Pedals are spring-loaded to return to normal position.)
7. Check rudder pedal shaker operation and adjust if necessary. (Refer to paragraph 6-231.)

FJ-48-2-51-121

6-235. APPROACH LIGHT SYSTEM.

6-236. The approach light system is completely automatic and is controlled by the angle-of-attack indicating system. A set of three colored lights, installed in the nose of the airplane, is actuated by switches, in the angle-of-attack indicator, which are preset at various angles of attack. An amber light indicates to the landing signal officer that the airplane is maintaining an optimum approach angle of attack; a red light indicates too low an approach angle of attack; a green light indicates too high an angle of attack for approach and suggests that the airplane will come in too slowly. Brilliance of the approach lights is controlled through the approach light relay which, in turn, is controlled by the EXTERIOR LIGHTS switch located forward of the power control lever on the left-hand forward vertical console. When the EXTERIOR LIGHTS switch is "OFF," the approach lights are bright; when the EXTERIOR LIGHTS switch is "ON," the approach lights are dim.

6-237. TROUBLE SHOOTING APPROACH LIGHT SYSTEM — AIRPLANES 139531 THROUGH 143593.

TEST EQUIPMENT: D-C voltmeter.
Ohmmeter.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND, EXTERIOR LIGHTS and LDG & APPROACH LIGHT & LANDING GEAR WARNING circuit breakers engaged.
Landing gear down and locked.
Arresting gear in down position.

Note

Airplane must be on jacks to allow arresting gear to come all the way down.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
AMBER APPROACH LIGHT WILL NOT ILLUMINATE—DAY OPERATION.			
Note Indicated angle of attack is 17 to 19 units.			
Defective bulb.			Replace defective bulb.
Defective approach light assembly.	Check between test point FLA and ground.	28 volts dc.	Replace defective approach light assembly.
		Zero volts.	Continue trouble shooting.
Defective approach light relay or defective associated wiring.	Check between test point FLB and ground.	28 volts dc.	Replace defective approach light relay or defective wire segment to last previous test point.
		Zero volts.	Continue trouble shooting.
	Check between test point FLC and ground.	28 volts dc.	Replace defective wire segment to last previous test point.
		Zero volts.	Continue trouble shooting.
Angle-of-attack indicator defective or out of adjustment or defective associated wiring.	Check between test point FLD and ground.	28 volts dc.	Replace or readjust angle-of-attack indicator (paragraphs 6-7, 6-8 and/or 6-238) or replace defective wire segment to last previous test point.
		Zero volts.	Continue trouble shooting.
Defective or misrigged arresting hook position down switch or defective associated wiring.	Check between test point FLE and ground.	28 volts dc.	Replace or rerig arresting hook position down switch (paragraph 3-330) or replace defective wire segment to last previous test point.
		Zero volts.	Continue trouble shooting.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
AMBER APPROACH LIGHT WILL NOT ILLUMINATE—DAY OPERATION. (Cont)			
Defective or misrigged arresting hook position down switch or defective associated wiring. (Cont)	Check between test point FLF and ground.	28 volts dc.	Replace defective wire segment to last previous test point.
		Zero volts.	Continue trouble shooting.
Defective L.G. DOWNLOCK relay, defective or misrigged landing gear downlock switch(es) or defective associated wiring.	Check between test point FLG and ground and between test points FLH and FLJ. **Note** Test point FLH is positive.	28 volts dc between test point FLG and ground and zero volts between test points FLH and FLJ.	Replace L.G. DOWNLOCK relay.
		Zero volts between test point FLG and ground.	Replace defective power wire.
		28 volts dc between test points FLH and FLJ.	Perform wire segment continuity check through landing gear downlock switches; rerig or replace defective landing gear downlock switch(es). (Refer to paragraphs 3-116 and 3-117.)

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND, EXTERIOR LIGHTS, LDG & APPROACH LIGHT & LANDING GEAR WARNING and ARREST HOOK circuit breakers engaged.
Landing gear down and locked.
Arresting gear up; momentarily toggle RUD. PED. SHAKER TEST AND APP LIGHT HOOK BY-PASS switch to the "APP LIGHT HOOK BY-PASS" position.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
AMBER APPROACH LIGHT WILL NOT ILLUMINATE—DAY OPERATION.			
Note ● Indicated angle of attack is 17 to 19 units. ● Repeat first four steps of isolation procedure, then proceed as follows.			
Defective HOOK BY PASS CONTROL relay, defective arresting hook control switch or defective associated wiring.	Check between test points FLK, FLL, FLM and ground.	28 volts dc.	Replace HOOK BY PASS CONTROL relay.
		Zero volts between test point FLK and ground.	Replace defective wire to relay.
		Zero volts between test point FLM and ground.	Replace defective d-c power wire or defective arresting hook control switch.
		Zero volts between test point FLL and ground.	Continue trouble shooting.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
AMBER APPROACH LIGHT WILL NOT ILLUMINATE—DAY OPERATION. (Cont)			
Defective L.G. DOWNLOCK relay, defective or misrigged landing gear downlock switch(es) or defective associated wiring.	Check between test point FLG and ground and between test points FLH and FLJ. **Note** Test point FLH is positive.	28 volts dc between test point FLG and ground and zero volts between test points FLH and FLJ.	Replace L.G. DOWNLOCK relay.
		Zero volts between test point FLG and ground.	Replace defective d-c power wire.
		28 volts dc between test points FLH and FLJ.	Perform wire segment continuity check through landing gear downlock switches. If necessary, rerig or replace defective landing gear downlock switch or switches. (Refer to paragraphs 3-116 and 3-117.)

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND, EXTERIOR LIGHTS, LDG & APPROACH LIGHT & LANDING GEAR WARNING and ARREST HOOK circuit breakers engaged.
EXTERIOR LIGHTS switch in "ON" position.
Landing gear down and locked.
Arresting gear up; momentarily toggle RUD. PED. SHAKER AND APP LIGHT HOOK BY-PASS switch to the "APP LIGHT HOOK BY-PASS" position.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
AMBER APPROACH LIGHT WILL NOT ILLUMINATE—NIGHT OPERATION ONLY.			
	Note Indicated angle of attack is 17 to 19 units.		
Defective approach light assembly.	Check between test point FLN and ground.	28 volts dc.	Replace approach light assembly.
		Zero volts.	Replace defective wire segment between test point FLN and approach light relay or replace defective approach light relay.
Defective approach light relay or defective associated wiring.	Check between test points FLB and FLP and ground.	28 volts dc.	Replace approach light relay.
		Zero volts between test point FLP and ground.	Continue trouble shooting.
Defective EXTERIOR LIGHTS switch or attached wiring.	Check between test point LEG and ground.	28 volts dc.	Replace defective EXTERIOR LIGHTS switch or attached wiring.
		Zero volts.	Continue trouble shooting.
Defective exterior lights control panel or associated wiring.	Check between test points FL1 and FL2.	Zero ohms.	Replace defective wire segment to approach light relay.
		Other than zero ohms.	Replace exterior lights control panel.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND, EXTERIOR LIGHTS, LDG & APPROACH LIGHT & LANDING GEAR WARNING circuit breakers engaged.
Landing gear down and locked.
Arresting gear in down position.

Note

Airplane must be on jacks to allow arresting gear to come all the way down.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY

RED APPROACH LIGHT WILL NOT ILLUMINATE—DAY OPERATION.

Note

Indicated angle of attack is 5 to 17 units.

The trouble shooting procedures used for the amber approach light may be utilized for this trouble by substituting the following test points:
Use test point FLR in place of test point FLA.
Use test point FLS in place of test point FLB.
Use test point FLT in place of test point FLC.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND, EXTERIOR LIGHTS, LDG & APPROACH LIGHT & LANDING GEAR WARNING and ARREST HOOK circuit breakers engaged.
Landing gear down and locked.
Arresting gear up; momentarily toggle RUD. PED. SHAKER AND APP LIGHT HOOK BY-PASS switch to the "APP LIGHT HOOK BY-PASS" position.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY

RED APPROACH LIGHT WILL NOT ILLUMINATE—DAY OPERATION.

Note

Indicated angle of attack is 5 to 17 units.

The trouble shooting procedures used for the amber approach light may be utilized for this trouble by substituting the following test points:
Use test point FLR in place of test point FLA.
Use test point FLS in place of test point FLB.
Use test point FLT in place of test point FLC.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND, EXTERIOR LIGHTS, LDG & APPROACH LIGHT & LANDING GEAR WARNING and ARREST HOOK circuit breakers engaged.
EXTERIOR LIGHTS switch in "ON" position.
Landing gear down and locked.
Arresting gear up; momentarily toggle RUD. PED. SHAKER AND APP LIGHT HOOK BY-PASS switch to the "APP LIGHT HOOK BY-PASS" position.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY

RED APPROACH LIGHT WILL NOT ILLUMINATE—NIGHT OPERATION ONLY.

Note

Indicated angle of attack is 5 to 17 units.

The trouble shooting procedures used for the amber approach light may be utilized for this trouble by substituting the following test points:
Use test point FLS in place of test point FLB.
Use test point FLU in place of test point FLN.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND, EXTERIOR LIGHTS, LDG & APPROACH LIGHT & LANDING GEAR WARNING circuit breakers engaged.
Landing gear down and locked.
Arresting gear in down position.

Note

Airplane must be on jacks to allow arresting gear to come all the way down.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
GREEN APPROACH LIGHT WILL NOT ILLUMINATE—DAY OPERATION.			

Note

Indicated angle of attack is 19 to 30 units.

The trouble shooting procedures used for the amber approach light may be utilized for this trouble by substituting the following test points:
Use test point FLW in place of test point FLA.
Use test point FLX in place of test point FLB.
Use test point FLY in place of test point FLC.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND, EXTERIOR LIGHTS, LDG & APPROACH LIGHT & LANDING GEAR WARNING and ARREST HOOK circuit breakers engaged.
Landing gear down and locked.
Arresting gear up; momentarily toggle RUD. PED. SHAKER AND APP LIGHT HOOK BY-PASS switch to the "APP LIGHT HOOK BY-PASS" position.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
GREEN APPROACH LIGHT WILL NOT ILLUMINATE—DAY OPERATION.			

Note

Indicated angle of attack is 19 to 30 units.

The trouble shooting procedures used for the amber approach light may be utilized for this trouble by substituting the following test points:
Use test point FLW in place of test point FLA.
Use test point FLX in place of test point FLB.
Use test point FLY in place of test point FLC.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND, EXTERIOR LIGHTS, LDG & APPROACH LIGHT & LANDING GEAR WARNING and ARREST HOOK circuit breakers engaged.
Landing gear down and locked.
Arresting gear up; momentarily toggle RUD. PED. SHAKER AND APP LIGHT HOOK BY-PASS switch to the "APP LIGHT HOOK BY-PASS" position.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
GREEN APPROACH LIGHT WILL NOT ILLUMINATE—NIGHT OPERATION ONLY.			

Note

Indicated angle of attack is 19 to 30 units.

The trouble shooting procedures used for the amber approach light may be utilized for this trouble by substituting the following test points:
Use test point FLX in place of test point FLB.
Use test point FLZ in place of test point FLN.

D-C POWER FAILURE.			
Circuit-breaker failure.	Check between test points PBR, PBT, PDM and ground.	28 volts dc.	Replace circuit breaker(s).
		Zero volts.	Refer to paragraph 8-61, Trouble Shooting D-C Power Distribution System.

6.23?A. TROUBLE SHOOTING APPROACH LIGHT SYSTEM AIRPLANES 143594m AND SUBSEQUENT.

TEST EQUIPMENT: D-C voltmeter.
Ohmmeter.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND, EXTERIOR LIGHTS and LDG & APPROACH LIGHT & LANDING GEAR WARNING circuit breakers engaged.
Landing gear down and locked.
Arresting gear in down position.

Note

Airplane must be on jacks to allow arresting gear to come all the way down.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
AMBER APPROACH LIGHT WILL NOT ILLUMINATE—DAY OPERATION.			
	Note Indicated angle of attack is 17 to 19 units.		
Defective bulb.			Replace defective bulb.
Defective approach light assembly.	Check between test point FLA and ground.	28 volts dc.	Replace defective approach light assembly.
		Zero volts.	Continue trouble shooting.
Defective approach light relay or defective associated wiring.	Check between test point FLB and ground.	28 volts dc.	Replace defective approach light relay or defective wire to last previous test point.
		Zero volts.	Continue trouble shooting.
Defective approach light control relay or associated wiring.	Check between test point FNA and ground.	28 volts dc.	Replace defective wire to last previous test point.
		Zero volts.	Continue trouble shooting.
	Check between test points FNB, FNC, FND and ground and between test points FND and FNE.	28 volts dc.	Replace defective wire between test points FNB and FNA.
		Zero volts at test point FNB or zero volts between test points FNE and FND.	Replace defective approach light control relay.
		Zero volts at test point FNC.	Perform wire segment continuity check to test point FNF and replace defective wires as required.
		Zero volts between test point FND and ground.	Continue trouble shooting.
Defective hook position down switch, defective L. G. DOWNLOCK relay or associated wiring.	Check between test point FLE and ground.	28 volts dc.	Replace or rig defective arresting hook position switch (paragraph 3-330) or replace defective wire to test point FND.
		Zero volts.	Continue trouble shooting.
	Check between test point FLF and ground.	28 volts dc.	Replace defective wire to last previous test point.
		Zero volts.	Continue trouble shooting.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
AMBER APPROACH LIGHT WILL NOT ILLUMINATE—DAY OPERATION. (Cont)			
Defective hook position down switch, defective L. G. DOWNLOCK relay or associated wiring. (Cont)	Check between test points FLG and ground and between test points FLH and FLJ. **Note** Test point FLH is positive.	28 volts dc between test point FLG and ground and zero volts between test points FLH and FLJ.	Replace L. G. DOWNLOCK relay.
		Zero volts between test point FLG and ground.	Replace defective power wire.
		28 volts dc between test points FLH and FLJ.	Perform wire segment continuity check through landing gear downlock switches; rig or replace defective landing gear downlock switches. (Refer to paragraphs 3-116 and 3-117.)
Angle-of-attack indicator defective or out of adjustment or defective associated wiring.	Check between test point FLD and ground.	28 volts dc.	Replace or adjust angle-of-attack indicator. (Refer to paragraphs 6-7, 6-8 and/or 6-238.)
		Zero volts.	Continue trouble shooting.
	Check between test point FLF and ground.	28 volts dc.	Perform wire segment continuity check to test point FLD and replace defective wire segment as required.
		Zero volts.	Refer to paragraph 8-14, Trouble Shooting D-C Power Supply System.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
LDG & APPROACH LIGHT & LANDING GEAR WARNING and ARREST HOOK circuit breakers engaged.
Landing gear down and locked.
Arresting gear up.
Momentarily toggle RUD. PED. SHAKER TEST AND APP LIGHT HOOK BY-PASS switch to the "APP LIGHT HOOK BY-PASS" position.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
AMBER APPROACH LIGHT WILL NOT ILLUMINATE—DAY OPERATION. (OPERATION IS SATISFACTORY WHEN NOT IN BY-PASS CONDITION.)			
	Note Indicated angle of attack is 17 to 19 units.		
Defective HOOK BY PASS CONTROL relay, defective arresting hook control handle switch or defective associated wiring.	Check between test points FLK, FLL and FLM.	28 volts dc.	Replace defective HOOK BY PASS CONTROL relay.
		Zero volts between test point FLK and ground.	Replace defective wire between test point FLK and HOOK BY PASS CONTROL relay.
		Zero volts between test point FLL and ground.	Trouble shoot L. G. DOWNLOCK relay. (Refer to the fifth probable cause of this chart.)
		Zero volts between test point FLM and ground.	Replace defective arresting hook control handle switch or replace defective wire between HOOK BY PASS CONTROL relay and the ARREST HOOK circuit breaker.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND, EXTERIOR LIGHTS, LDG & APPROACH LIGHT & LANDING GEAR WARNING and ARREST HOOK circuit breakers engaged.
EXTERIOR LIGHTS switch in "ON" position.
Landing gear down and locked.
Arresting gear up; momentarily toggle RUD. PED. SHAKER TEST AND APP LIGHT HOOK BY-PASS switch to the "APP LIGHT HOOK BY-PASS" position.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
AMBER APPROACH LIGHT WILL NOT ILLUMINATE—NIGHT OPERATION ONLY—DAY OPERATION SATISFACTORY.			
	Note Indicated angle of attack is 17 to 19 units.		
Defective approach light assembly.	Check between test point FLN and ground.	28 volts dc.	Replace approach light assembly.
		Zero volts.	Replace defective wire segment between test point FLN and approach light relay or replace defective approach light relay.
Defective approach light relay or defective associated wiring.	Check between test points FLB and FLP and ground.	28 volts dc.	Replace approach light relay.
		Zero volts between test point FLP and ground.	Continue trouble shooting.
Defective EXTERIOR LIGHTS switch or attached wiring.	Check between test point LEG and ground.	28 volts dc.	Replace defective EXTERIOR LIGHTS switch or attached wiring.
		Zero volts.	Continue trouble shooting.
Defective exterior lights control panel or associated wiring.	Check between test points FL1 and FL2.	Zero ohms.	Replace defective wire segment to approach light relay.
		Other than zero ohms.	Replace exterior lights control panel.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND, EXTERIOR LIGHTS and LDG & APPROACH LIGHT & LANDING GEAR WARNING circuit breakers engaged.
Landing gear down and locked.
Arresting gear in down position.

Note

Airplane must be on jacks to allow arresting gear to come all the way down.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
RED APPROACH LIGHT WILL NOT ILLUMINATE—DAY OPERATION.			
	Note Indicated angle of attack is 5 to 17 units.		
	The trouble shooting procedures used for the amber approach light may be utilized for this trouble by substituting the following test points: Use test point FLR in place of test point FLA. Use test point FLS in place of test point FLB. Use test point FNG in place of test point FNA.		

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND, EXTERIOR LIGHTS, LDG & APPROACH LIGHT & LANDING GEAR WARNING and ARREST HOOK circuit breakers engaged.
Landing gear down and locked.
Arresting gear up; momentarily toggle RUD. PED. SHAKER TEST AND APP LIGHT HOOK BY-PASS switch to the "APP LIGHT HOOK BY-PASS" position.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY

RED APPROACH LIGHT WILL NOT ILLUMINATE—DAY OPERATION.

Note

Indicated angle of attack is 5 to 17 units.

The trouble shooting procedures used for the amber approach light may be utilized for this trouble by substituting the following test points:

Use test point FNH in place of test point FNB.
Use test point FNJ in place of test point FNC.
Use test point FNK in place of test point FNF.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND, EXTERIOR LIGHTS, LDG & APPROACH LIGHT & LANDING GEAR WARNING and ARREST HOOK circuit breakers engaged.
EXTERIOR LIGHTS switch in "ON" position.
Landing gear down and locked.
Arresting gear up; momentarily toggle RUD. PED. SHAKER TEST AND APP LIGHT HOOK BY-PASS switch to the "APP LIGHT HOOK BY-PASS" position.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY

RED APPROACH LIGHT WILL NOT ILLUMINATE—NIGHT OPERATION ONLY.

Note

Indicated angle of attack is 5 to 17 units.

The trouble shooting procedures used for the amber approach light may be utilized for this trouble by substituting the following test points:

Use test point FLS in place of test point FLB.
Use test point FLU in place of test point FLN.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND, EXTERIOR LIGHTS and LDG & APPROACH LIGHT & LANDING GEAR WARNING circuit breakers engaged.
Landing gear down and locked.
Arresting gear in down position.

Note

Airplane must be on jacks to allow arresting gear to come all the way down.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY

GREEN APPROACH LIGHT WILL NOT ILLUMINATE—DAY OPERATION.

Note

Indicated angle of attack is 19 to 30 units.

The trouble shooting procedures used for the amber approach light may be utilized for this trouble by substituting the following test points:

Use test point FLW in place of test point FLA.
Use test point FLX in place of test point FLB.
Use test point FNL in place of test point FNA.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND, EXTERIOR LIGHTS, LDG & APPROACH LIGHT & LANDING GEAR WARNING and ARREST HOOK circuit breakers engaged.
Landing gear down and locked.
Arresting gear up; momentarily toggle RUD. PED. SHAKER TEST AND APP LIGHT HOOK BY-PASS switch to the "APP LIGHT HOOK BY-PASS" position.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY

GREEN APPROACH LIGHT WILL NOT ILLUMINATE—DAY OPERATION.

Note

Indicated angle of attack is 19 to 30 units.

The trouble shooting procedures used for the amber approach light may be utilized for this trouble by substituting the following test points:

Use test point FNM in place of test point FNB.
Use test point FNN in place of test point FNC.
Use test point FNP in place of test point FNF.

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
ANGLE OF ATTACK IND, EXTERIOR LIGHTS, LDG & APPROACH LIGHT & LANDING GEAR WARNING and ARREST HOOK circuit breakers engaged.
EXTERIOR LIGHTS switch in "ON" position.
Landing gear down and locked.
Arresting gear up; momentarily toggle RUD. PED. SHAKER TEST AND APP LIGHT HOOK BY-PASS switch to the "APP LIGHT HOOK BY-PASS" position.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY

GREEN APPROACH LIGHT WILL NOT ILLUMINATE—NIGHT OPERATION ONLY.

Note

Indicated angle of attack is 19 to 30 units.

The trouble shooting procedures used for the amber approach light may be utilized for this trouble by substituting the following test points:

Use test point FLX in place of test point FLB.
Use test point FLZ in place of test point FLN.

D-C POWER FAILURE.

Circuit-breaker failure.	Check between test points PBR, PBT, PDM and ground.	28 volts dc.	Replace circuit breaker(s).
		Zero volts.	Refer to paragraph 8-61, Trouble Shooting D-C Power Distribution System.

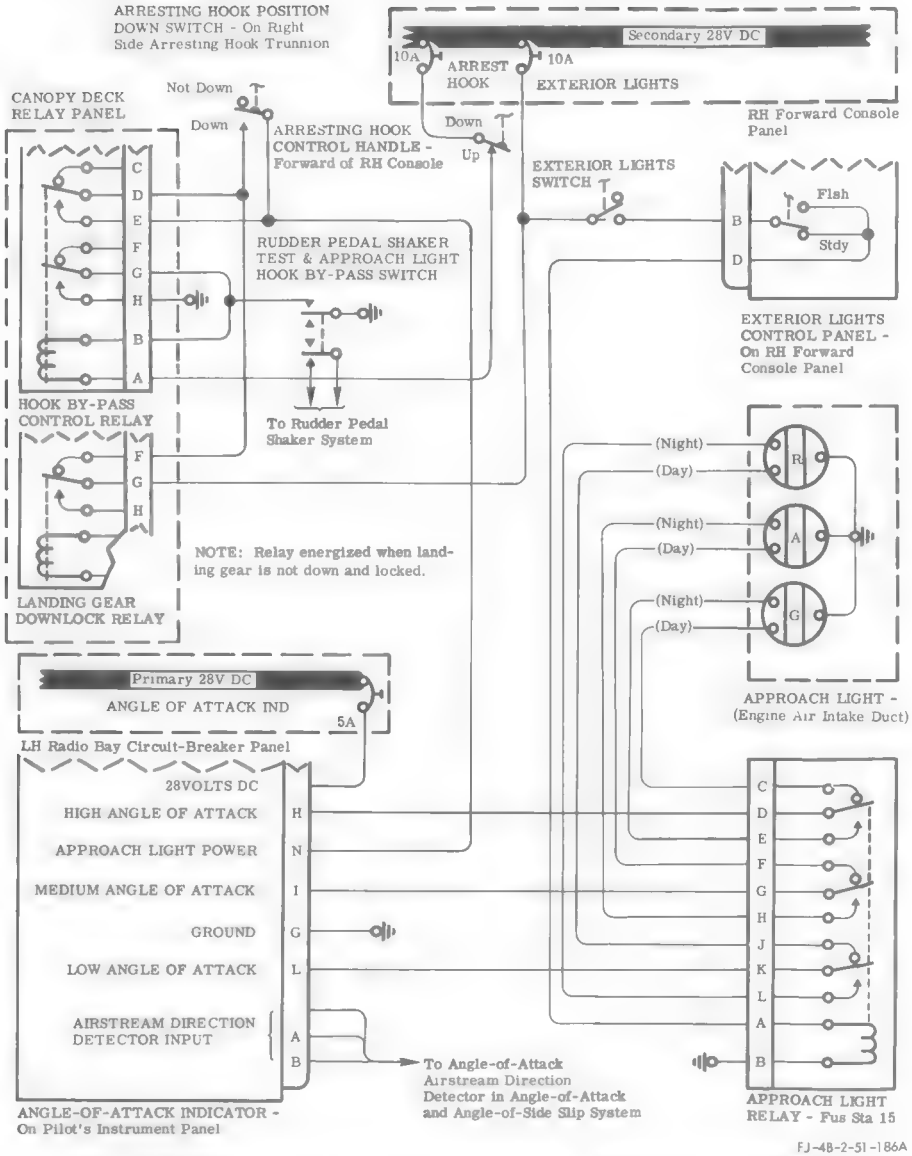


Figure No. 6-47. Approach Light System Schematic—Airplanes 139531i through 143593i

6-238. ADJUSTING APPROACH LIGHT AND APPROACH INDEXER SYSTEMS.*

Note

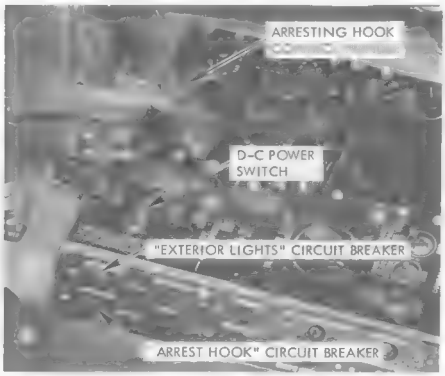
*Approach indexer system installed on airplanes 143594m and subsequent.

- The landing gear must be down and locked.
- Three men are necessary to adjust the approach lights and the approach indexer: one in the cockpit, one at the approach lights and one at the angle-of-attack airstream direction detector.

1 Place the d-c power switch, located on the right-hand forward console, in the "OFF" position.

2 Engage the following circuit breakers: The ANGLE-OF-ATTACK IND located on the left-hand radio bay circuit-breaker panel, the EXTERIOR LIGHTS and the ARREST HOOK, located on the right-hand forward console panel.

3 Make sure the arresting hook control handle, located forward of the right-hand console, is in the UP position. This step is not applicable when adjusting the approach indexer.



4 Make sure the EXTERIOR LIGHTS switch, located on the left-hand forward vertical console panel forward of the power control lever, and the instrument lights rheostat (INSTRUMENTS) located on the right-hand console panel are in the "OFF" position.

5 Connect a source of external power to the airplane.

6 Place the RUD. PED. SHAKER TEST AND APP LIGHT HOOK BY-PASS switch, located on the test switch panel, in the "APP LIGHT HOOK BY-PASS" position. This step is not applicable when adjusting the approach indexer.

FJ-48-2-51-126A

7 Remove protective cover from airstream direction detector probe and slowly rotate probe to its extreme clockwise position. The indicator should read -5 units. Rotate the probe to its extreme counterclockwise position. The indicator should read 30 units.

Note The approach lights, approach indexer and angle-of-attack indicator should be observed closely while the probe is rotated.

8 The dial indication change-over points for the approach lights and the approach indexer should be as follows:

DIAL INDICATION	APPROACH LIGHTS	APPROACH INDEXER LIGHTS
-5 TO 16.5	RED	BOTTOM
16.5 TO 17.0	RED	BOTTOM AND CENTER
17.0 TO 19.0	AMBER	CENTER
19.0 TO 19.5	GREEN	CENTER AND TOP
19.5 TO 30.0	GREEN	TOP

FJ-48-2-51-127A

If the proper approach light and the proper approach indexer light(s) are illuminated at their correct dial indication, the cams in the angle-of-attack indicator are correctly adjusted; if not, proceed as follows:

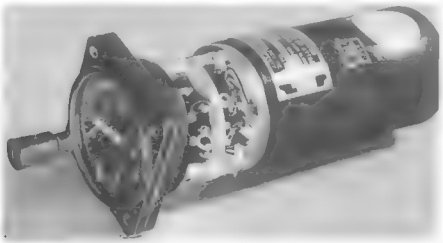
9 Disengage ANGLE-OF-ATTACK IND, ARREST HOOK and EXTERIOR LIGHTS circuit breakers.

10 Remove angle-of-attack indicator from the instrument panel. (Refer to paragraph 6-7.)

11 Cut safety wire and remove three screws from rear of indicator.

12 Remove cylindrical cover to expose five cam adjustments.

Note The cams are designated No. 1 through No. 5, with No. 1 being nearest the dial.



13 Reconnect the connector to the rear of the angle-of-attack indicator.

14 Engage ANGLE OF ATTACK IND, ARREST HOOK and EXTERIOR LIGHTS circuit breakers.

FJ-48-2-51-128A

15 Toggle RUD. PED. SHAKER AND APP LIGHT HOOK BY-PASS switch to "APP LIGHT HOOK BY-PASS" position. This step not applicable when adjusting the approach indexer.

Note The holding circuit to the HOOK BY-PASS CONTROL relay is broken whenever the ARREST HOOK circuit breaker is disengaged, thus the RUD. PED. SHAKER AND APP LIGHT HOOK BY-PASS switch must again be toggled to the "APP LIGHT HOOK BY-PASS" position to re-energize the by-pass relay.

16 Rotate the angle-of-attack probe to its upper limit stop. The approach light should be red and the bottom approach indexer light will still be illuminated.

17 Rotate the angle-of-attack probe counterclockwise until the indicator reads 16.5 units.

18 Rotate cam No. 3 around the camshaft until the center approach indexer light illuminates.

Note The red approach light and the bottom approach indexer light will still be illuminated at this point.

Note The switching point of the cam is adjusted by inserting a 1/8-inch dowel pin or drill rod in the radial hole on one side of the cam and using it as a lever to rotate the cam relative to the camshaft. When switch is actuated, remove lever; cam will again grip camshaft and move with the shaft.

19 Rotate the angle-of-attack probe counterclockwise until the indicator reads 17.0 units.

20 Adjust cam No. 5 around the camshaft until the amber approach light illuminates.

FJ-48-2-51-129A

Note The red approach light and the bottom approach indexer light will extinguish at this point.

21 Rotate the angle-of-attack probe counterclockwise until the indicator reads 19.0 units.

22 Rotate cam No. 4 around the camshaft until the green approach light and the top approach indexer light illuminates.

Note The center approach indexer light will remain illuminated at this point.

23 Rotate the angle-of-attack probe counterclockwise until the indicator reads 19.5 units.

24 Rotate cam No. 2 around the camshaft until the center approach indexer light is extinguished.

Note The green approach light and the top approach indexer light will remain illuminated throughout the remainder of the angle-of-attack probe travel.

25 Place the EXTERIOR LIGHTS switch and the instrument lights rheostat (INSTRUMENTS) in the "ON" position to energize the approach light relay and the auxiliary warning light dimming relay respectively.

26 Rotate the angle-of-attack probe from one limit stop to the other to check the dimming circuits of the approach lights and the approach indexer lights.

27 Return the EXTERIOR LIGHTS switch and the instrument lights rheostat (INSTRUMENTS) to the "OFF" position.

28 Remove source of external power.

29 Remove connector from rear of angle-of-attack indicator.

30 Replace cover on angle-of-attack indicator and install three cover securing screws. Safety screws with AN995F32 wire.

31 Install angle-of-attack indicator. (Refer to paragraph 6-8.)

FJ-4B-2-51-183A

6-239. APPROACH LIGHTS.

6-240. The approach lights are installed in the nose of the airplane so as to be clearly visible to the landing signal officer. The assembly consists of three separate lights: one with a red lens, one with an amber lens and one with a green lens. Each of the three lamps contains two separate filaments — one for day brilliance and one for night brilliance. The approach lights are individually lighted by means of two cam switches in the angle-of-attack indicator.

6-240A. APPROACH INDEXER.

6-240B. The approach indexer is installed on airplanes 143594m and subsequent to provide the pilot with an illuminated indication of the landing approach angle of attack. The approach indexer is mounted on the left-hand center windshield defrost line and uses components of the angle-of-attack indicating system to receive the required electrical signals. The approach indexer is composed of a light assembly and an anti-glare hood. The face of the indexer hinges aft and up to facilitate replacement of one or more of the three bulbs which it contains. Indication is provided through three apertures on the face of the unit. A "V"-shaped aperture at the top of the indexer indicates a low angle of attack. The "O"-shaped aperture in the center specifies an optimum or medium angle of attack. The inverted "V"-shaped aperture at the bottom of the unit means the angle of attack of the airplane is high. When the top and center lamps are both illuminated at the same time, the angle of attack is medium low. If the center and bottom lamps are both illuminated at the same time, a medium high angle of attack is denoted. All indications are red. The approach indexer will be operable whenever power is applied to the airplane and the landing gear is down and locked, regardless of the position of the arresting gear. The indexer is mounted in the pilot's forward line of vision so that approach indexer indications and the runway, mirror or landing signal officer may be observed at the same time. All three bulbs in the indexer are capable of being dimmed whenever the instrument lights rheostat (INSTRUMENTS) is in any position but "OFF." The dimming circuit is routed through the auxiliary warning light dimming relay and three 500-ohm resistors (one for each of the three bulbs).

6-240C. TROUBLE SHOOTING APPROACH INDEXER SYSTEM — AIRPLANES 143594m AND SUBSEQUENT.

TEST EQUIPMENT: D-C voltmeter.
Ohmmeter.
Jumper wire. (Refer to GENERAL INFORMATION, Section VIII, for fabrication.)

SYSTEM CONDITIONS: 28-volt d-c power applied to airplane.
EXTERIOR LIGHTS, INST. LIGHTS and ANGLE OF ATTACK IND circuit breakers engaged.
Landing gear down and locked.
Angle-of-attack indicator plug disconnected.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
ONE OR MORE APPROACH INDEXER LIGHTS INOPERATIVE.			
Angle-of-attack indicator defective or out of adjustment, defective auxiliary warning light dimming relay, defective approach indexer bulb(s) or defective associated wiring. **Note** Instrument lights rheostat (INSTRUMENTS) is in the "OFF" position.	Connect jumper to test point FAB and, in turn, to test points FMA, FMB and FMC.	With jumper between test points FAB and FMA, the high (top) approach indexer light is illuminated.	Replace or readjust angle-of-attack indicator if 28 volts dc is available at test point FLD. If 28 volts dc is not available at test point FLD, refer to trouble shooting the L.G. DOWNLOCK relay and associated circuitry in paragraph 6-237A, Trouble Shooting Approach Light System—Airplanes 143594m and Subsequent.
		With jumper between test points FAB and FMA, the high (top) approach indexer light is not illuminated.	Perform wire segment continuity check between test points FMA and FMD and replace defective wire, defective auxiliary warning light dimming relay or defective approach indexer bulb as required.
		With jumper between test points FAB and FMB, the medium (center) approach indexer light is illuminated.	Replace or readjust angle-of-attack indicator if 28 volts dc is available at test point FLD. If 28 volts dc is not available at test point FLD, refer to trouble shooting the landing gear downlock relay and associated circuitry in paragraph 6-237A, Trouble Shooting Approach Light System—Airplanes 143594m and Subsequent.
		With jumper between test points FAB and FMB, the medium (center) approach indexer light is not illuminated.	Perform wire segment continuity check between test points FMB and FME and replace defective wire, defective auxiliary warning light dimming relay or defective approach indexer bulb as required.

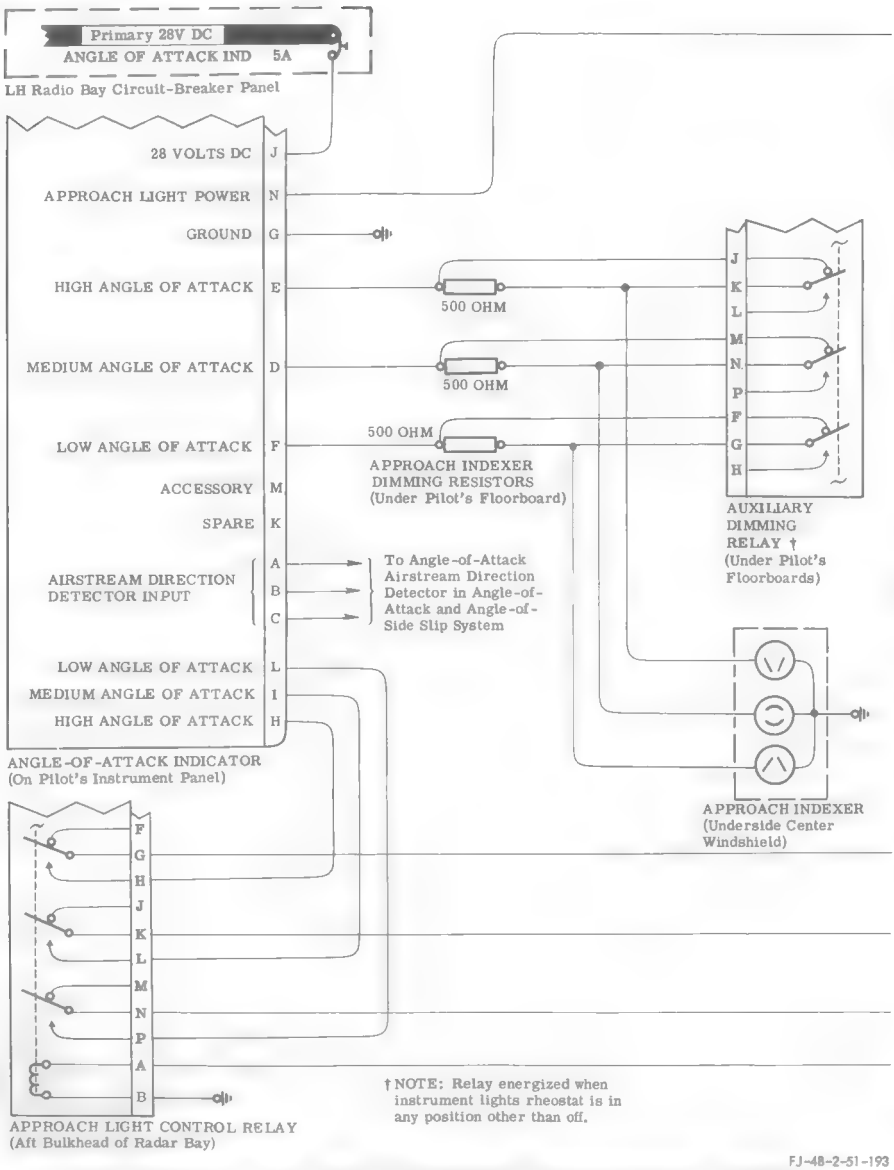


Figure No. 6-47A. Approach Light and Approach Indexer Systems Schematic—Airplanes 143594m and Subsequent (Sheet 1)

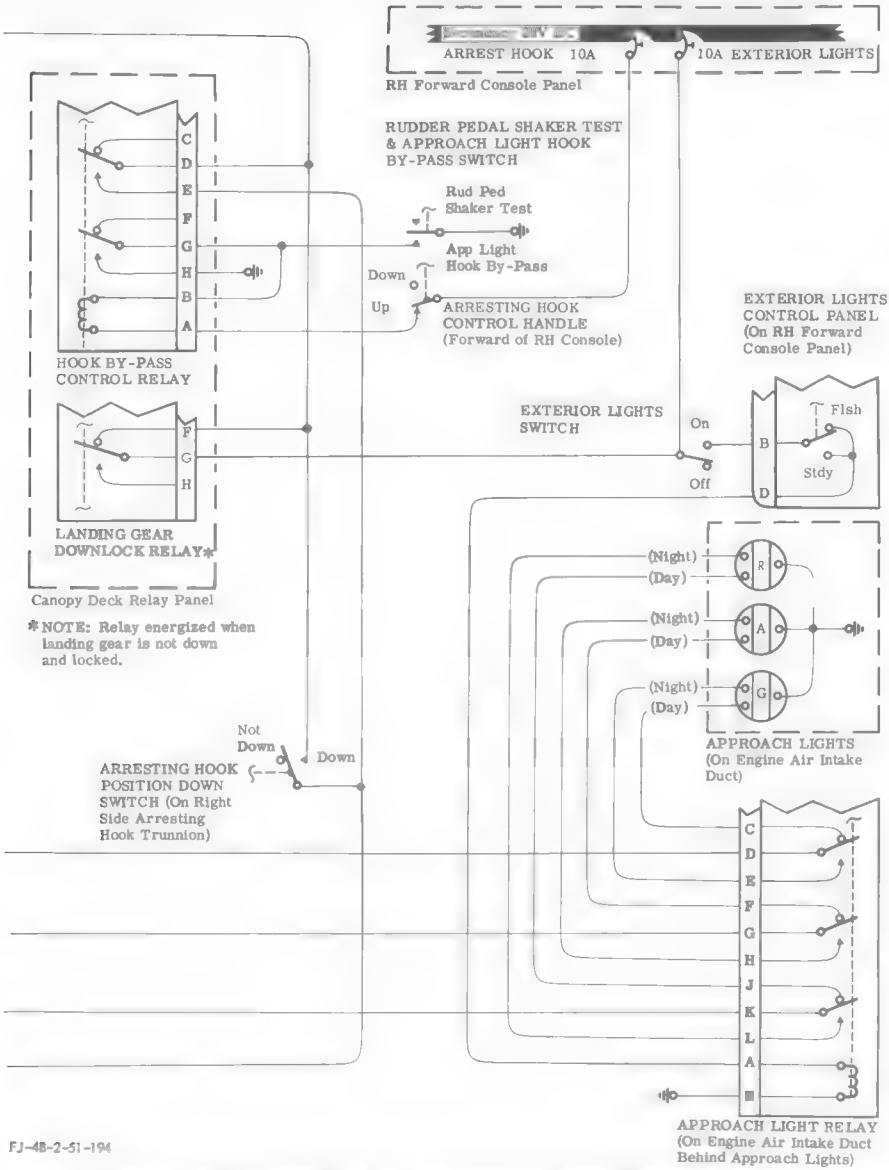


Figure No. 6-47A. Approach Light and Approach Indexer Systems Schematic—Airplanes 143594m and Subsequent (Sheet 2)

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
ONE OR MORE APPROACH INDEXER LIGHTS INOPERATIVE. (Cont)			
Angle-of-attack indicator defective or out of adjustment, defective auxiliary warning light dimming relay, defective approach indexer bulb(s) or defective associated wiring. **Note** Instrument lights rheostat (INSTRUMENTS) is in the "OFF" position. (Cont)	Connect jumper to test point FAB and, in turn, to test points FMA, FMB and FMC. (Cont)	With jumper between test points FAB and FMC, the low (bottom) approach indexer light is illuminated.	Replace or readjust angle-of-attack indicator if 28 volts dc is available at test point FLD. If 28 volts dc is not available at test point FLD, refer to trouble shooting the landing gear downlock relay and associated circuitry in paragraph 6-237A, Trouble Shooting Approach Light System—Airplanes 143594m and Subsequent.
		With jumper between test points FAB and FMC, the low (bottom) approach indexer light is not illuminated.	Perform wire segment continuity check between test points FMC and FMF and replace defective wire, defective auxiliary warning light dimming relay or defective approach indexer bulb as required.
Angle-of-attack indicator defective or out of adjustment, defective auxiliary warning light dimming relay, defective approach indexer bulb(s) or defective associated wiring. **Note** Instrument lights rheostat (INSTRUMENTS) is in any position other than "OFF."	Use preceding isolation procedure for this probable cause.	Use preceding meter readings for this probable cause.	Substitute warning light dimming resistor(s) in place of warning light dimming relay and use preceding remedies for this probable cause.
D-C POWER FAILURE.			
Circuit-breaker failure.	Check between test points PBR, PBT, PDM and ground.	28 volts dc.	Replace circuit breaker(s).
		Zero volts.	Refer to paragraph 8-61, Trouble Shooting D-C Power Distribution System.

EXHAUST TEMPERATURE INDICATING SYSTEM

6-241. EXHAUST TEMPERATURE INDICATING SYSTEM.

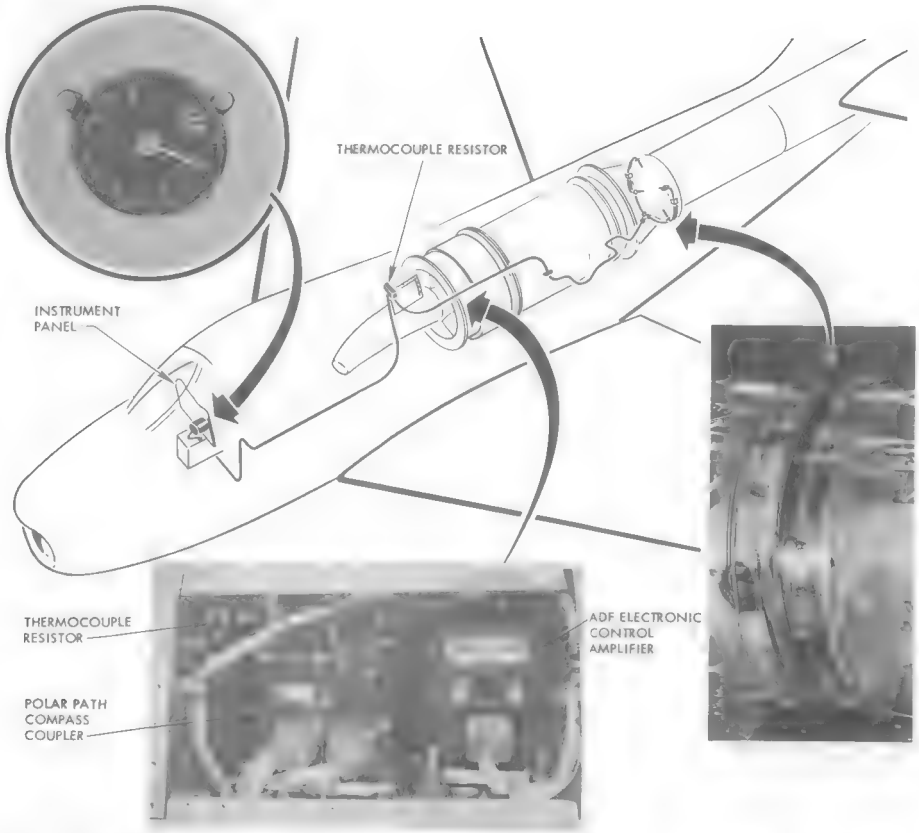
6-242. The exhaust temperature indicating system (figure 6-48) measures the temperatures of the exhaust gases from the engine by converting heat energy into electrical energy and then measuring the electrical energy produced. The system consists of four thermocouples supplied with the engine, thermocouple leads, a variable resistor and an exhaust temperature indicator. The four thermocouples are wired in parallel to indicate the average exhaust temperature of the engine. The thermocouple lead circuit resistance is set for 8 ($\pm$0.05) ohms and should be checked each time the engine is replaced, a new section of lead is installed or a different thermocouple is installed. The variable resistor is located in the upper part of the fuselage radio and electrical equipment compartment on the left-hand side of the airplane.

6-243. TROUBLE SHOOTING EXHAUST TEMPERATURE INDICATING SYSTEM.

Note

The following trouble shooting chart should be used only when a Jetcal Analyzer and Jetcal test equipment are not available.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
NO INDICATION.		
Broken thermocouple lead.	Visually check the thermocouple circuit from the engine thermocouples to the indicator.	Replace lead as necessary.
Loose or dirty connections.	Visually check the thermocouple lead connections at the indicator. Check thermocouple lead connections at the resistor, at the thermocouple disconnect on the engine and at the thermocouple harness on the engine.	Reconnect or replace leads or replace indicator as necessary. Reconnect or replace leads, disconnects or segment connector on bottom of engine.
Defective indicator.		Replace indicator.
VARIABLE READING.		
Broken thermocouple lead, crossed lead or a short in lead.	Perform an operational check of exhaust temperature indicating system. (Refer to paragraph 6-244.) Visually check the thermocouple circuit from the engine thermocouples to the indicator.	Replace lead as necessary; correct crossed leads or shorts.
Defective indicator.		Replace indicator.
INDICATOR READS BACKWARDS.		
Reversed leads at indicator.	Check that white lead (chromel) is connected to the "+" connecting stud and that the green lead (alumel) is connected to the "—" connecting stud.	Reverse leads if necessary.



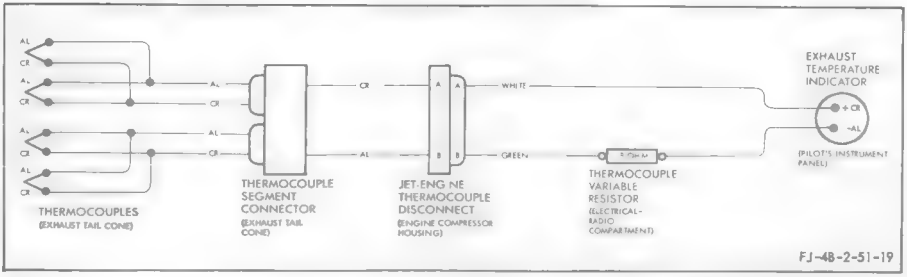


Figure No. 6-48. Exhaust Temperature Indicating System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
HIGH INDICATION.		
Defective indicator.		Replace indicator.
EXTREME OSCILLATION OF INDICATOR POINTER OR POINTER STICKS OR TRACKS ERRATICALLY. INTERMITTENT INDICATION.		
Defective indicator.		Replace indicator.
LOW INDICATION.		
Broken thermocouple lead, crossed lead or a short in lead.	Perform an operational check of exhaust temperature indicating system. (Refer to paragraph 6-244.) Visually check the thermocouple circuit from the engine thermocouples to the indicator.	Replace lead as necessary; correct crossed leads or shorts.
Defective indicator.		Replace indicator.

6-244. OPERATIONAL CHECK OF EXHAUST TEMPERATURE INDICATING SYSTEM.

6-245. To perform an operational check of the exhaust temperature indicating system, proceed as follows:

a. Start the engine (figure 1-15) and slowly accelerate the engine to 100% rpm.

b. Record the peak exhaust gas temperature reached at 100% rpm and simultaneously start timing the engine operation.

Note

If a Jetcal Analyzer is available (paragraph 6-246), maintain 100% rpm according to the tachometer indication on the Jetcal Analyzer instrument panel.

c. Record the exhaust gas temperature every 30 seconds until the engine has stabilized at some one temperature for a 3-minute period.

CAUTION

During any acceleration above idle speed, the exhaust gas temperature may attain a maximum of 800°C for a maximum of 10 seconds. Thirty seconds after the start of the acceleration, the temperature *must* drop to 690°C or below. If these limits are exceeded one time only, perform a hot section inspection on the engine. If engine exceeds 106% rpm, shut down and subject engine to a hot section inspection. If engine exceeds 108% rpm, or if temperature limits are exceeded more than once, shut down and submit engine to an overhaul facility.

d. The exhaust gas temperature should stabilize between 640°C and 650°C at 100% rpm after 10 minutes.

Note

Exhaust gas temperature limits are for an ambient temperature of 37.8°C (100°F), a Navy hot day. When performing an operational check in lower ambient temperatures, the maximum limits will be correspondingly lower. Refer to exhaust gas temperature versus ambient temperature chart (figure 5-10).

e. If the operational check is not satisfactory, perform a functional check of the exhaust temperature indicating system utilizing the Jetcal Analyzer and Jetcal test equipment. (Refer to paragraph 6-246.) If the exhaust temperature indicating system proves to be functionally accurate, trouble shoot the engine. (Refer to paragraph 5-3.)

6-246. FUNCTIONAL CHECK OF EXHAUST TEMPERATURE INDICATING SYSTEM.

6-247. A functional check of the exhaust temperature indicating system should be performed on all airplanes that have an engine or system component replaced, or whenever exhaust temperature difficulties have been encountered. The function of the system is checked with a Jetcal Analyzer (B & H Instrument Co., Inc., Model BH112J) with B & H accessories for checking system resistance, exhaust temperature indications, thermocouple calibration and for isolating system errors. All necessary check and heater cables are included with the Analyzer assembly, but the following additional equipment is necessary to complete a system check:

BH378	Probe assembly (4 probes required)
BH361-12	Junction box
BH473	Check cable adapter
C11A	Portable power cart

Essentially, the functional check of the system is accomplished by artificially heating the engine's thermocouples in the exhaust cone (without running the engine or disconnecting the system) to an engine test temperature. With the thermocouples hot, this temperature is registered on the exhaust temperature indicator. At the

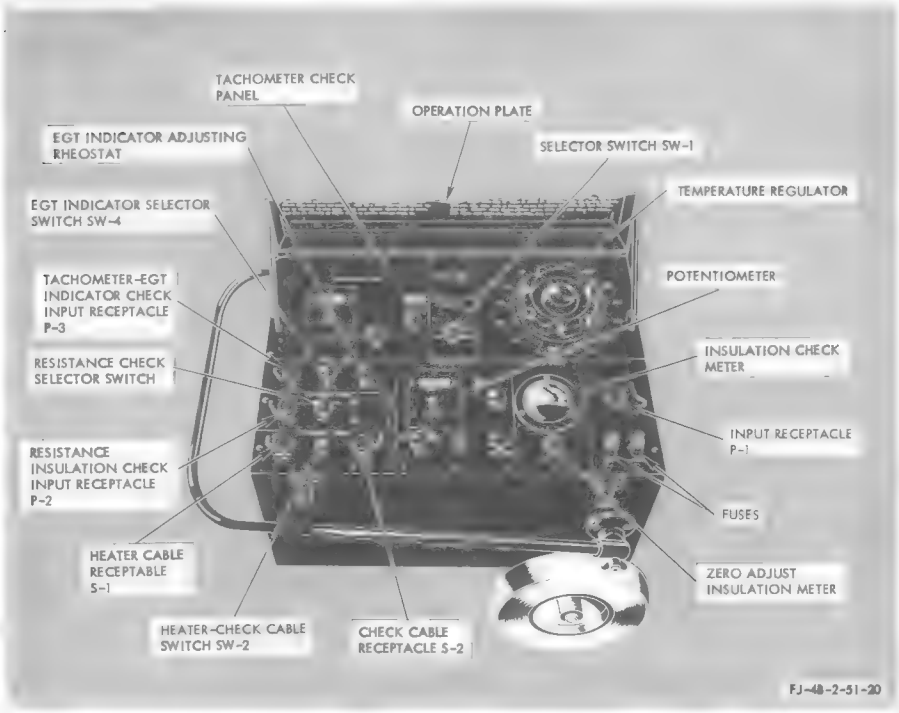


Figure No. 6-49. Jetcal Analyzer Instrument Compartment

same time, thermocouples embedded in the heater probes are picking up and registering this same temperature on the potentiometer in the Jetcal Analyzer. The temperature on the exhaust temperature indicator should agree, within a specified tolerance, with the Jetcal Analyzer reading. To perform an exhaust temperature indicating system functional check, proceed as follows:

a. On the Jetcal Analyzer instrument panel (figure 6-49), set TEMP REGULATOR to "0" and set selector switch SW-1 to "OFF."

b. Connect one end of the power inlet cable to input receptacle P-1 and the other end to a 95- to 135-volt, 50- to 400-cycle, a-c power source (C11A portable power cart may be used).

CAUTION

The Jetcal Analyzer should be grounded with the pigtail lead of the power inlet cable.

c. Connect junction box to heater cable and heater cable to Jetcal Analyzer at heater cable receptacle S-1.

d. Connect one heater probe (BH378) to junction box.

e. Place selector switch SW-1 in "T/C" position and place switch SW-2 in "HEATER CABLE" position.

f. Set up potentiometer panel (figure 6-50) as follows: (1) Place selector switch SW-6 in "MECH. ZERO" position; zero GALVO-1 by turning MECH. ZERO knob. (2) Hold selector switch SW-6 in "ELEC. ZERO" position; zero GALVO-1 by turning ELEC. ZERO knob R-3. (3) Place selector switch SW-6 in "RANGE" position to take temperature readings; adjust temperature selector knob until GALVO-1 reads "0." Temperature reading appears on °C scale.

Note

Check potentiometer and temperature reading by turning temperature selector knob until GALVO-1 reads "0" with the probe heater cold. Temperature should indicate existing ambient temperature.

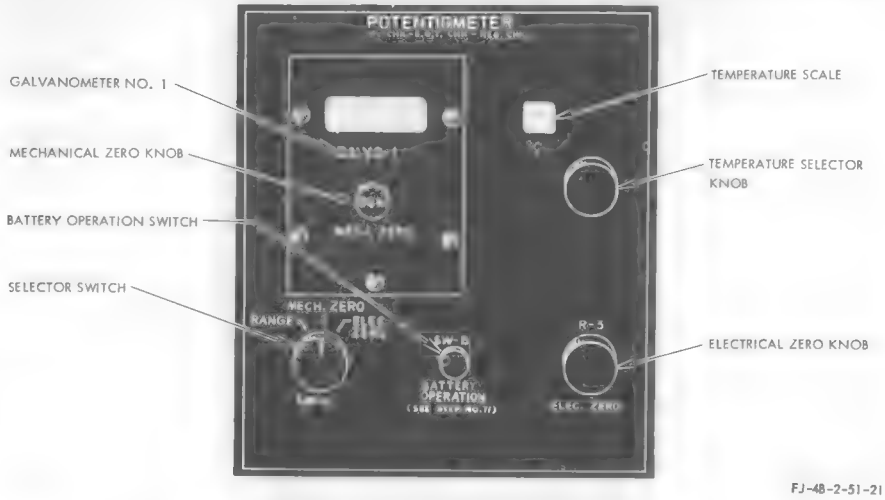


Figure No. 6-50. Jetcal Potentiometer Panel

g. Turn TEMP REGULATOR to maximum and bring the one heater probe to engine test temperature of 650°C.

Note

To make rapid adjustments of heater probe temperatures, turn TEMP REGULATOR to maximum until heater probe comes to test temperature; then, turn TEMP REGULATOR back to approximately 80 volts (for 650°C test temperature). Watch galvanometer needle closely to see which way it drifts. Note regulator setting. Turn regulator well above or below setting for a few seconds; then, return regulator to a setting which is one or two volts above or below original setting (whichever is required). Repeat procedure until probe stabilizes at test temperature.

h. To eliminate inoperative thermocouple error in the circuit, place the one hot heater probe over each of the airplane thermocouples (figure 6-51), one at a time. The airplane's exhaust temperature indicator should show a temperature rise as each thermocouple is checked.

i. Place selector switch SW-6 in "MECH. ZERO" position and turn TEMP REGULATOR back to "0."

j. Place one heater probe over each thermocouple in the system and connect each probe to the junction box. (See figure 6-51.)

k. Place selector switch SW-6 in "RANGE" position.

l. Turn TEMP REGULATOR to maximum and bring heater probes to engine test temperature of 650°C. Allow sufficient time for the airplane thermocouples to stabilize.

Note

The time necessary to heat and to stabilize probe and thermocouple temperatures will depend upon line voltage, ambient temperatures and air currents. If testing in high wind or cold weather, place a cover over exhaust cone.

m. After the temperature of the airplane thermocouples has stabilized, note exhaust temperature indicator reading and compare with the temperature on Jetcal Analyzer indicator. The difference in temperatures is the indicated error of the exhaust temperature indicating system and should not exceed ±10°C tolerance. If this tolerance is exceeded, perform trouble isolation procedures. (Refer to paragraph 6-250.)

WARNING

If exhaust temperature indicator is 25°C low (or greater), replace the engine.

Note

Do not adjust airplane's exhaust temperature indicator to agree with Jetcal temperature indication.

n. If the exhaust temperature indicating system is within a tolerance of ±10°C, place selector switch SW-6 in "MECH. ZERO" position and rotate TEMP REGULATOR to "0."

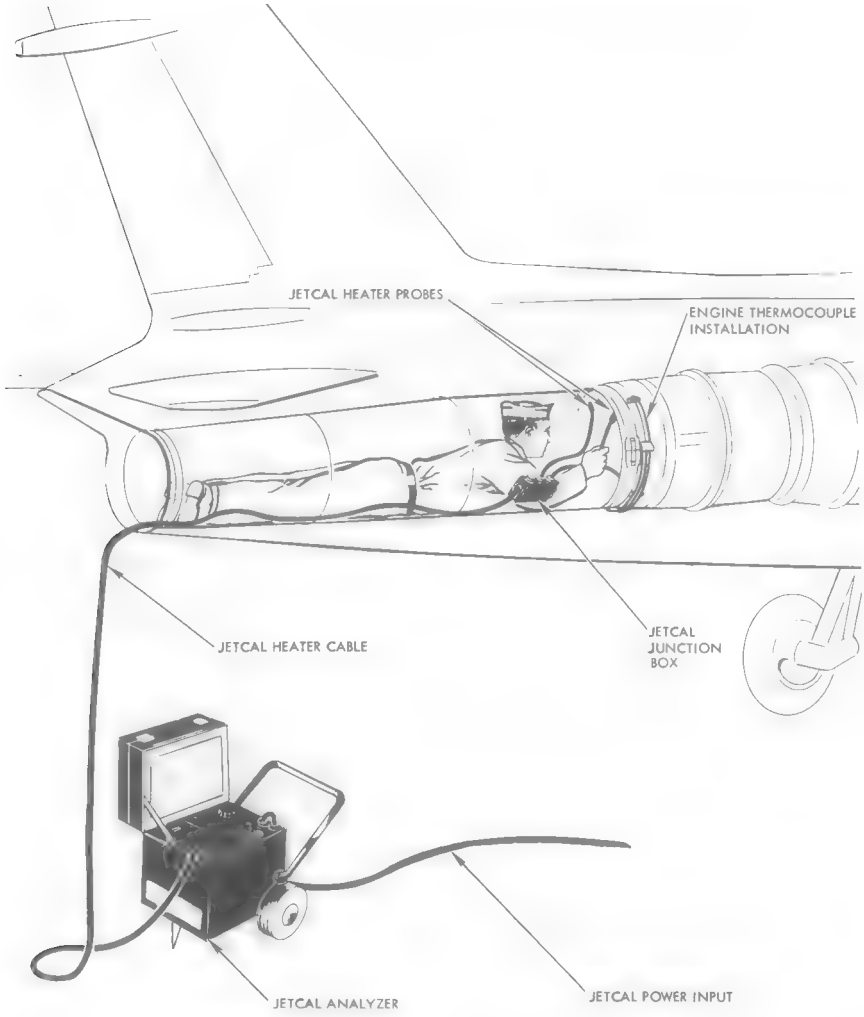


Figure No. 6-51. Installation of Jetcal Heater Probes on Airplane Thermocouples

o. Place selector switch SW-1 in "OFF" position.

p. Remove power supply, heater cables, heater probes and junction box from airplane and from Jetcal Analyzer.

6-248. CALIBRATING EXHAUST TEMPERATURE INDICATING SYSTEM.

6-249. The exhaust temperature indicating system should be calibrated with the Jetcal Analyzer following engine replacement or replacement of system components. Calibration of the engine thermocouples or indicator should be accomplished following system disconnection, or if the exhaust temperature indicator is out of tolerance with the tester during a functional check. To calibrate the exhaust temperature indicating system, proceed as follows:

a. Check that Jetcal calibration is valid.

b. Check engine thermocouple calibration. (Refer to paragraph 6-257.)

c. Check exhaust temperature indicator using Jetcal Analyzer. (Refer to paragraph 6-252.) Indicator should have zero error at engine test temperature of 650°C.

d. Disconnect exhaust temperature indicating system at the exhaust temperature indicator and connect thermocouple leads to the resistance check adapter. (See figure 6-52.)

e. Connect resistance check adapter RES to instrument cable and instrument cable to input receptacle P-2.

f. With power connected to Jetcal (steps a. and b., paragraph 6-246), place selector switch SW-6 in "MECH. ZERO" position. Zero GALVO-1 by turning MECH. ZERO knob.

g. Turn switch SW-3 to 8 (±0.05) ohms resistance.

h. Place selector switch SW-1 in "RES" position.

i. Place selector switch SW-6 in "RANGE" position.

j. Check that the airplane exhaust temperature indicating circuit resistance is 8 (±0.05) ohms (GALVO-1 will be zeroed). One division of GALVO-1 equals approximately 0.05 ohms.

k. If resistance is not 8 (±0.05) ohms, adjust the variable resistor (paragraph 6-259) until GALVO-1 indicates "0."

CAUTION

Be sure the engine thermocouples are at ambient temperature before checking the resistance of the circuit.

l. Place selector switch SW-6 in "MECH. ZERO" position and place selector switch SW-1 in "OFF" position.

m. Remove check cable and adapter.

n. Connect the thermocouple leads to the exhaust temperature indicator and torque lead nuts to 23 (±2) inch-pounds.

o. Install Jetcal thermocouple heater probes and adjust test temperature to 650°C. (Refer to paragraph 6-246.)

p. Compensate for Jetcal calibration if any.

q. The temperature of the exhaust temperature indicator must agree with Jetcal temperature within the allowable tolerance of ±10°C. If the tolerance is exceeded, perform trouble isolation procedures. (Refer to paragraph 6-250.)

6-250. EXHAUST TEMPERATURE INDICATING SYSTEM TROUBLE ISOLATION PROCEDURES.

6-251. When the exhaust temperature indicating system is found to be in error by more than ±10°C at the test temperature of 650°C (by a functional test using the Jetcal Analyzer), one of the following conditions may be the source of trouble. The test procedures described should be accomplished in attempting to isolate the trouble.

6-252. DEFECTIVE EXHAUST TEMPERATURE INDICATOR. The most probable source of trouble in the exhaust temperature circuit is a defective exhaust temperature indicator. To check the indicator with the Jetcal Analyzer, proceed as follows:

a. Loosen the tension screw at the lower right of the exhaust temperature indicator and remove the instrument from the panel.

b. Disconnect the chromel (white) lead from the +CR stud on the indicator.

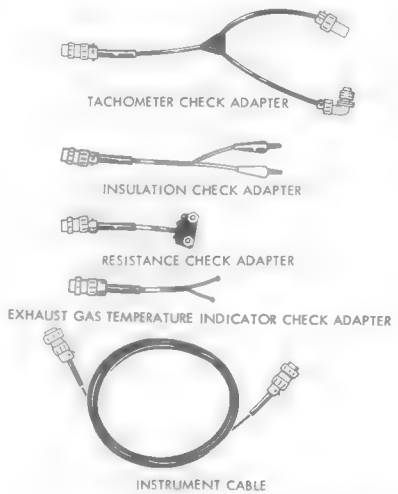


Figure No. 6-52. Jetcal Instrument Cable and Adapters

c. Place selector switch SW-1 in "OFF" position.

d. Connect power inlet cable to Jetcal at receptacle P-1 and to power supply.

e. Connect EGT indicator check adapter (figure 6-52) to instrument cable and instrument cable to input receptacle P-3 in Jetcal Analyzer.

f. Connect EGT indicator check adapter to the proper posts on the exhaust temperature indicator.

Note

The indicator must be kept in the normal operating position in order to have an accurate check.

g. Place selector switch SW-1 in "EGT" position.

h. Place selector switch SW-6 in "MECH. ZERO" position. Zero GALVO-1 by turning MECH. ZERO knob.

i. Hold selector switch SW-6 in "ELEC. ZERO" position. Zero GALVO-1 by turning ELEC. ZERO knob R-3.

j. Set potentiometer to indicate test temperature of 650°C.

k. Place selector switch SW-4 to 8 (±0.05) ohms resistance.

l. Place selector switch SW-6 in "RANGE" position.

m. Adjust ADJ RHEO R-1 rheostat until GALVO-1 reads "0."

n. The difference between the readings of the potentiometer and the exhaust temperature indicator is the error of the indicator. There should be zero error at 650°C.

o. Place selector switch SW-6 in "MECH. ZERO" position and place selector switch SW-1 in "OFF" position.

p. Remove cables and adapter from Jetcal and exhaust temperature indicator.

q. Reconnect the chromel lead to the +CR stud on the indicator.

r. Position the exhaust temperature indicator in the instrument panel and secure by tightening the lower right tension screw.

6-253. DEFECTIVE OR INOPERATIVE THERMOCOUPLE. If one or more thermocouples are defective, or if there is faulty wiring from a thermocouple, current from the faulty thermocouple may be grounded out and never reach the indicator. To check the thermocouples, refer to paragraph 6-246 and perform steps a. through i.

Note

Be sure to place the hot heater probe over each thermocouple, *one* at a time.

Check grounding of thermocouples by performing the insulation resistance check. (Refer to paragraph 6-255.)

6-254. RESISTANCE OF CIRCUIT OUT OF TOLERANCE. If the resistance of the thermocouple circuit is not adjusted to 8 (±0.05) ohms, incorrect indications will result. The resistance may be checked by using the Jetcal Analyzer and test equipment. (Refer to paragraph 6-259.) If the resistance needs adjusting, adjust at the variable spool resistor located in the upper compartment of the fuselage radio and instrument compartment.

6-255. SHORTS TO GROUND (INSULATION RESISTANCE CHECK). To check for a short to ground in the thermocouple circuit, proceed as follows:

a. Connect instrument cable to input receptacle P-2 on Jetcal Analyzer.

b. Connect insulation check adapter INSUL (figure 6-52) to instrument cable.

c. Place selector switch SW-1 in "INSUL" position.

d. Short leads of the insulation check adapter and adjust ZERO ADJ R-2 rheostat until insulation check meter reads "0."

e. Unshort leads of insulation check adapter and place one lead of adapter on bare thermocouple wire in the airplane and the other to airplane ground (structure).

f. Insulation check meter should indicate 100,000 ohms or more.

6-256. SHORTS BETWEEN LEADS (INSULATION RESISTANCE CHECK). To check for shorts between leads using the Jetcal insulation resistance check, proceed as follows:

a. Loosen the tension screw at the lower right side of the exhaust temperature indicator and remove the instrument from the panel.

b. Disconnect both thermocouple leads from the instrument.

c. Remove left-hand engine access door.

d. Disconnect the thermocouple circuit at the engine thermocouple disconnect.

e. Refer to paragraph 6-255 and perform steps a. through d.

f. Place the insulation check adapter leads (figure 6-52) on both of the thermocouple leads (at the instrument end or the engine end).

g. Insulation check meter should read 100,000 ohms or more.

h. Reconnect the thermocouple circuit at the engine thermocouple disconnect.

i. Replace engine access door.

j. Reconnect the thermocouple leads to the exhaust temperature indicator.

k. Position the exhaust temperature indicator in the instrument panel and secure by tightening tension screw in the lower right-hand corner.

6-257. ENGINE THERMOCOUPLE ERROR. The error of the thermocouples, or engine thermocouple harness, may be checked on the potentiometer in the Jetcal Analyzer which eliminates resistance errors. To check for thermocouple error, proceed as follows:

a. Loosen the tension screw at the lower right of the exhaust temperature indicator and remove the instrument from the panel.

b. Disconnect both thermocouple leads from the exhaust temperature indicator.

c. Connect both thermocouple leads to the proper terminals of the tabbing switch box.

d. Connect short jumper leads of switch box to proper terminals on the exhaust temperature indicator.

e. Connect check cable to switch box and Jetcal Analyzer.

f. Turn switch SW-5 on switch box to "JETCAL" position.

g. Place heater probes on all engine thermocouples and heat to test temperature of 650°C. (Refer to paragraph 6-246.)

h. Place switch SW-2 in "HEATER CABLE" position and read temperature of heater probes on potentiometer.

i. Place switch SW-2 in "CHECK CABLE" position and read temperature of engine thermocouples on potentiometer. The algebraic difference between the two readings is the thermocouple harness error. Replace thermocouples if error exceeds ±10°C. (Refer to paragraph 6-267.)

j. Double check instrument reading while heater probes are on engine thermocouples by placing switch SW-5 on switch box in "EGT" position. This puts the exhaust temperature indicator in the circuit and the instrument reading is compared to the Jetcal potentiometer reading.

k. Place selector switch SW-6 in "MECH. ZERO" position and rotate TEMP REGULATOR to "0."

l. Place selector switch SW-1 in "OFF" position.

m. Remove heater probes and cables from airplane and Jetcal equipment.

n. Reconnect thermocouple leads to the exhaust temperature indicator.

o. Position the exhaust temperature indicator in the panel and secure by tightening tension screw in the lower right-hand corner of the instrument.

6-258. TRANSPOSITIONS (CROSSING OF LEAD WIRES). An odd number of crossed leads in the thermocouple circuit can easily be detected since the exhaust temperature indicator will read backwards if this situation exists when the thermocouples are heated. An even number of crossed leads may be detected by using one of the following methods:

a. The simplest method of checking transposition is with an ohmmeter. Chromel has approximately two and one-half times the resistance of alumel wire. Therefore, the chromel thermocouple lead should have approximately two and one-half times the resistance of the alumel thermocouple lead. To have a transposition, some of the opposite wires must have been inserted into that lead and, therefore, will show a different resistance than a two and one-half to one ratio.

b. If possible, make a normal thermocouple test with the Jetcal Analyzer on a hot engine (shortly after the engine has been running). When the engine has cooled to ambient temperature, make another test. If there is a transposition, the difference in the readings will probably be large because of the thermals that were present in the first test. In other words, one of the crossed junctions had been hot and the other cross junction cold which induces thermal in the system.

c. A magnet may be used to check for transposition. Chromel (+) is nonmagnetic; alumel (−) is magnetic. Disconnect each terminal and check with the magnet to see if the connections are correct.

d. Touch each terminal in the system with a hot soldering iron tip or one of the Jetcal probes. A temperature rise or fall on the exhaust temperature indicator will indicate a crossed terminal at that point.

6-259. ADJUSTING EXHAUST TEMPERATURE INDICATING SYSTEM CIRCUIT RESISTANCE.

CAUTION

Make certain the temperature of the engine thermocouples is close to ambient temperature before checking and adjusting the resistance of the circuit.

Note

This complete procedure should be performed when adjusting the circuit without a Jetcal Analyzer unit. If a Jetcal Analyzer is being used to calibrate the system, perform only those procedure steps necessary to gain access to the resistor spools and to decrease the circuit resistance; no Wheatstone bridge is necessary with a Jetcal Analyzer.

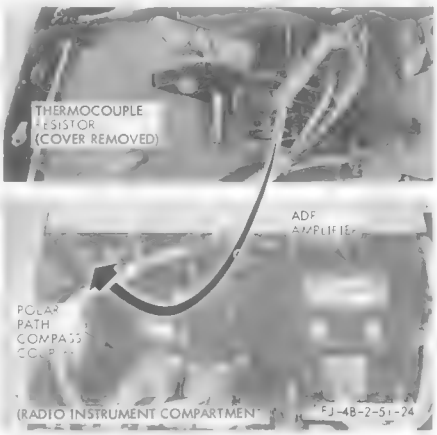


Figure No. 6-53. Thermocouple Variable Resistor

To adjust the exhaust temperature indicating system resistance, proceed as follows:

a. Loosen the tension screw at the lower right side of the exhaust temperature indicator and remove the instrument from the panel.

b. Disconnect the thermocouple lead from the positive stud (+CR) on rear of exhaust temperature indicator and connect it to the negative lead so that both leads are connected to the negative stud (−AL).

c. Remove the radio and electrical compartment access door located on the left-hand side of the airplane.

d. Remove cover from the thermocouple resistor. (See figure 6-53.)

e. Attach a Wheatstone bridge between open end of resistance spool and open end of thermocouple lead and measure total resistance of circuit. (See figure 6-54.)

f. Unwind one coil at a time from resistance spool, measuring the resistance of the thermocouple circuit for each turn removed until resistance measured is slightly more than 8 ohms.

CAUTION

Do not remove too much resistance so as to reduce the total resistance to less than 8 ohms. This wire has a resistance of approximately 0.657 ohm per foot.

g. Gradually remove resistance from the spool until the total resistance is 8 (±0.05) ohms.

h. Wrap the wire several turns around the nonconducting post provided for that purpose (to prevent unwinding); then, solder the resistance wire to the terminal.

i. Secure the ends of the resistance wire on spare spool using several wraps of vinyl tape.

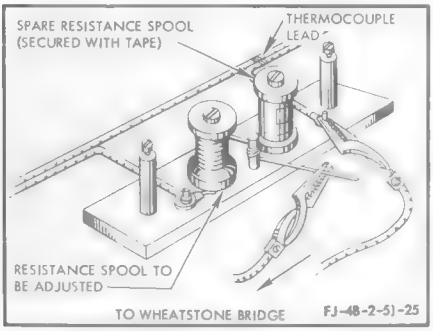


Figure No. 6-54. Adjustment of Thermocouple Resistance

j. Connect the Wheatstone bridge between the indicator terminals of the thermocouple leads and check the resistance of the entire circuit. The resistance should be 8 (±0.05) ohms.

k. Reconnect thermocouple leads to the correct terminals on indicator.

l. Position exhaust temperature indicator in the panel and secure by tightening tension screw in the lower right-hand corner of the instrument.

m. Replace the cover on resistor and secure with safety wire.

n. Replace and secure access door.

6-260. THERMOCOUPLE LEADS.

6-261. Two thermocouple leads (R88-L-0515-250), one chromel and the other alumel, connect the thermocouples to the indicator. The white chromel lead is positive and connects the chromel of the thermocouples directly to the positive terminal on the indicator. The green alumel lead is negative and is routed from the alumel of the thermocouples to the variable thermocouple resistor (connected in series with it) and from there to the negative terminal on the indicator. The chromel-alumel leads are insulated duplex wire with a resistance of 7 ohms per 50 feet.

6-262. EXHAUST TEMPERATURE INDICATOR.

6-263. The exhaust temperature indicator is basically a millivoltmeter which measures the electromotive force produced when the thermocouples become heated. This value is indicated in degrees of temperature on a dial calibrated from 0°C to 10°C (times 100). The indicator consists of a moving coil, mounted in the field of a permanent magnet, an adjusting resistance spool, a neutralizer and a compensating spring. The two thermocouple leads are attached to either end of the moving coil which rotates as current flows through it. The indicating pointer, attached to the coil, moves through an angle proportional to the current flowing through the coil. Since the basic function of the indicator is to measure the difference between the cold junction (at the indicator) and the hot junction (at the thermocouples), the scale is calibrated to indicate exhaust temperature and not just the differential temperature. In order to compensate for changes in cockpit temperature which would affect the indication, a compensating spring assembly is incorporated in the indicator. This compensating spring, which is attached to the moving coil, consists of two dissimilar metals having a widely different temperature coefficient of expansion. This spring will unwind when heated and will wind up when cooled, controlling the motion of the moving coil. The indication will be the true exhaust gas temperature, regardless of the temperature at the cold junction. On the rear of the instrument case, the positive and negative connecting posts are of different diameters and are marked +CR and −AL to ensure proper connection of the two leads to the indicator. An adjustment screw is provided on the rear of the instrument case to adjust the temperature indicator for zero tolerance.

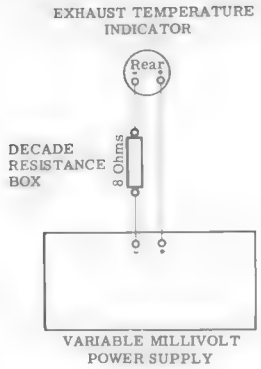


Figure No. 6-55. Schematic for Adjusting Exhaust Temperature Indicator

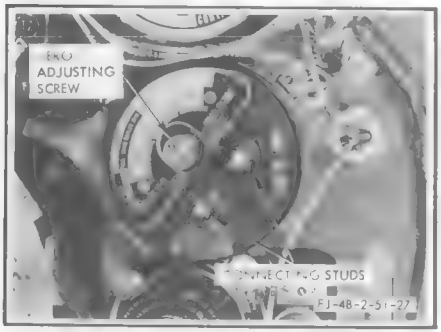


Figure No. 6-56. Exhaust Temperature Indicator Zero Adjustment Screw

6-264. ADJUSTING EXHAUST TEMPERATURE INDICATOR. The exhaust temperature indicator should be adjusted for zero tolerance at an engine test temperature of 650°C. To adjust the indicator, proceed as follows:

a. Loosen tension screw at lower right side of exhaust temperature indicator and remove indicator from instrument panel.

b. Remove thermocouple leads from indicator.

c. Subject the indicator to a constant room temperature for at least one hour.

d. Connect the indicator to a standard millivolt power supply in series with 8 ohms resistance. (See figure 6-55.)

e. Apply the correct millivoltage according to the ambient room temperature. Refer to the following chart:

MILLIVOLTAGE PER TEST POINT
650°C PER AMBIENT TEMPERATURE

ROOM TEMPERATURE °C	°F	MILLIVOLTS APPLIED
10	50	26.62
15	59	26.42
20	68	26.22
25	77	26.02
30	86	25.82
35	95	25.62

Note

For intermediate ambient temperatures, subtract 0.04 millivolt for each °C above the next lower ambient temperature listed.

f. With indicator held in normal operating position and with correct millivoltage applied, set pointer at 650°C by turning zero adjustment screw on rear of indicator case. (See figure 6-56.)

Note

Tap indicator lightly while making the adjustment to reduce friction within the case.

g. Seal zero adjusting screw by applying a 1/16-inch wide white marker across adjusting slot.

h. Remove power supply and resistance.

i. Reconnect thermocouple leads to proper terminals on the indicator.

j. Position the exhaust temperature indicator in the instrument panel and secure by tightening the tension screw at the lower right of the indicator.

6-265. EXHAUST TEMPERATURE INDICATOR THERMOCOUPLES.

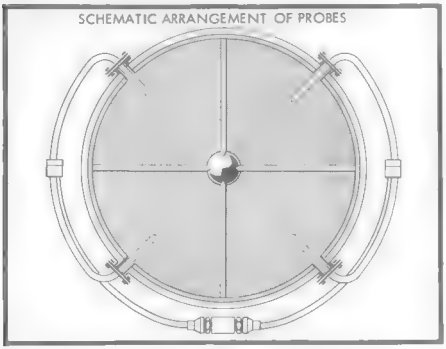
6-266. The four exhaust temperature thermocouples are supplied with the engine and are installed around the circumference of the exhaust tail cone. Each thermocouple consists of two wires of dissimilar metals, one of chromel and one of alumel, which are formed into a cable and joined together at one end. The junction end is inserted into the exhaust gases. When the thermocouples become heated, an electromotive force is produced which is transmitted to the indicator by two thermocouple leads.

6-267. REMOVING AND INSTALLING EXHAUST TEMPERATURE INDICATOR THERMOCOUPLES.

REMOVING

1 Remove fuselage aft section. (Refer to paragraph 2-6.)

2 Disconnect the two thermocouple probe segments from the connector at bottom of engine.



3 At each thermocouple probe, remove two bolts and nuts that secure thermocouple flange to mounting boss.

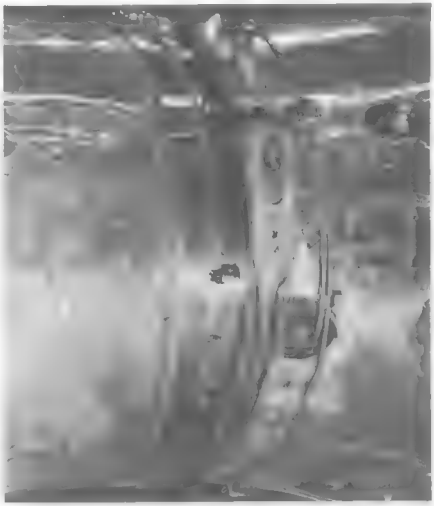
4 Carefully lift the two left-hand thermocouple probes from the wells and remove complete left-hand segment from the engine; repeat procedure for right-hand thermocouples.

INSTALLING

1 Position left-hand thermocouple segment and carefully insert the two thermocouple probes in the wells.

2 Secure each thermocouple flange to mounting boss with two bolts and nuts.

3 Install right-hand thermocouple segment and probes.



4 Connect thermocouple segments to connector on bottom of engine.

5 Perform an operational check of exhaust temperature indicating system. (Refer to paragraph 6-244.)

6 Install fuselage aft section. (Refer to paragraph 2-6.)

FJ-48-2-51-29

ENGINE FIRE DETECTOR SYSTEM

6-268. ENGINE FIRE DETECTOR SYSTEM.

6-269. The airplane is equipped with two independent, continuous-type fire detector systems (figure 6-57). One system is in the engine compressor section and the other system is in the engine burner section. The systems consist basically of two continuous loops of fire detector sensing elements, a control unit for each sensing element, located in the left-hand radio bay directly above the No. 2 inverter, and two warning lights mounted on the instrument panel. A fire detector test switch (FIRE DET), integral with the two systems, is provided. The test switch disconnects one end of both sensing circuits from the control unit and connects the circuits to ground. The switch checks for continuity of the sensing element, operation of the control units and continuity of the airplane wiring. The continuous-type fire detector systems utilize lengths of tubing (approximately 0.090-inch outside diameter) connected in series and appropriately routed within the airplane. These tubes, which are the system sensing elements, contain a semiconductor ceramic material surrounding two center wire electrical conductors. The electrical resistance of the ceramic material varies inversely with temperature so that, when the element is exposed to elevated temperatures, the electrical resistance between the center wire and the outer tube decreases. This resistance change is monitored by the control unit which provides an alarm when the resistance decreases to a predetermined value. The warning lights and test switch are mounted on the instrument panel in the cockpit. The lights are the illuminated plate-type with the words "FIRE BURNER" on the face of the light for the aft circuit and the words "FIRE COMPRESSOR" on the face of the light for the forward circuit. Both lights are red when illuminated. A test switch, located on the right-hand console, is used to test the bulbs in the fire warning lights.

6-270. FUNCTION OF ENGINE FIRE DETECTOR SYSTEM.

6-271. The system sensing elements are routed in the cooling air stream and over points of air exit. Consequently, any excessive heat will be immediately detected due to the airflow and/or flame going through and out of the compartment past the elements. In addition, elements are routed in or near sump areas which might contain flammable liquid drainage so that fires in these regions will be readily detected. The fire detector system, located in the engine burner compartment (aft of the fire wall), indicates an overheat condition upon initial warning and a fire condition if the warning cannot be eliminated by engine power reduction. No overheat detection is provided for the compressor compartment (forward of the fire wall) because of the relatively low fire resistance of the section, the combustible sources present and the resultant need for immediate pilot action. The temperature settings for the systems are as follows:

a. In the forward circuit (compressor compartment), the compressor circuit consists of 40 feet of sensing element (figure 6-57) set to alarm when any one foot of element length is exposed to a temperature of 737°F (±35°F).

b. In the aft circuit (burner compartment), the burner circuit consists of 50 feet of sensing element (figure 6-57) set to alarm when any one foot of element length is exposed to a temperature of 737°F (±35 F).

The fire resistant detector elements and connections create a system considered to be safe from a failure due to fire and, therefore, capable of indicating a "fire out" condition as well as a "fire" condition.

6-272. TROUBLE SHOOTING ENGINE FIRE DETECTOR SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
FIRE DETECTOR SYSTEM GIVES A CONSTANT WARNING. (WARNING LIGHT BURNS CONTINUALLY WHILE AIRCRAFT BATTERY OR EXTERNAL POWER SOURCE IS CONNECTED.)		
Sensing element or aircraft wire shorted.	**Perform the sensing element resistance tests as given in paragraph 6-273. If resistance is normal and light is still on, the relay in the control unit is faulty.**	**Replace relay.**
	Disconnect sensing elements from circuit and check resistance of circuit. If short still exists, it is in the fire zone wiring. Check wiring carefully for chafing, abrasions or crushed wires.	**Replace faulty wiring.**

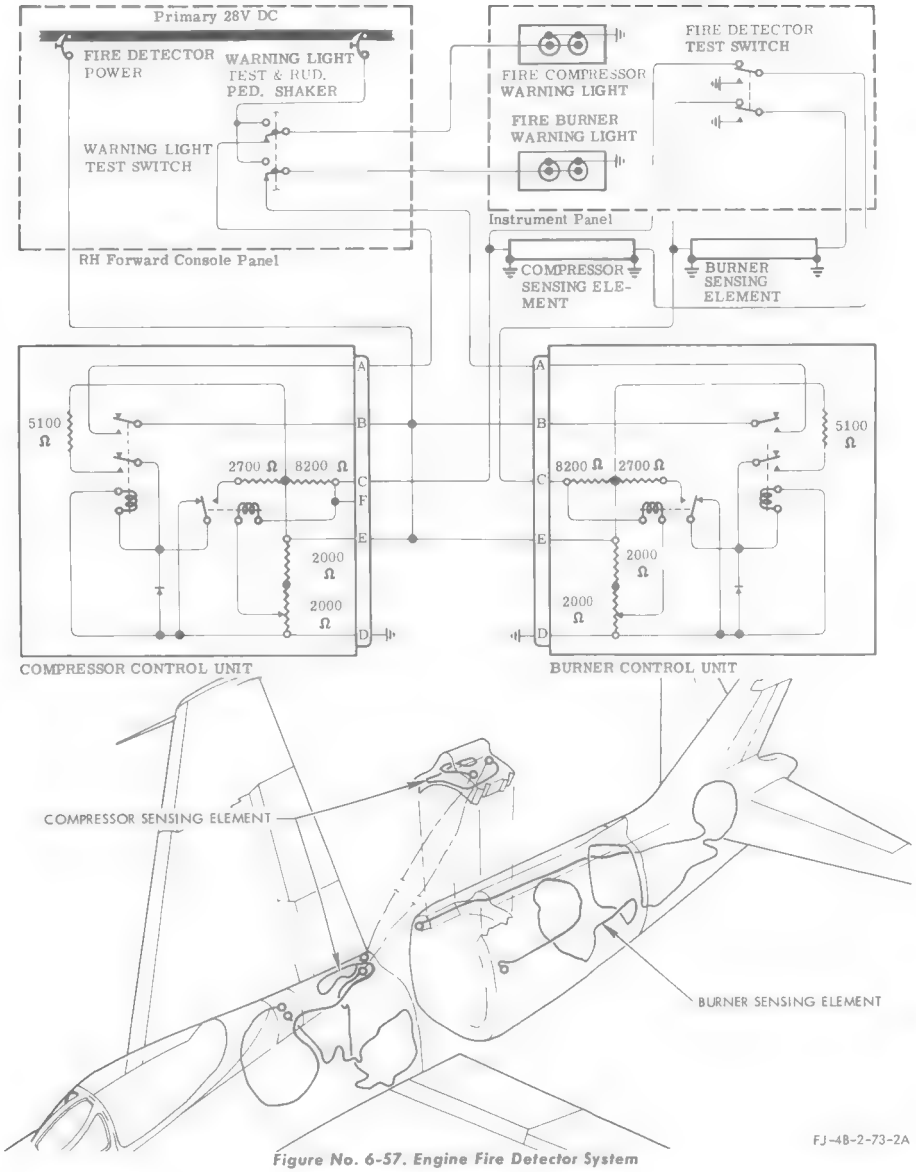


Figure No. 6-57. Engine Fire Detector System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
FIRE DETECTOR SYSTEM GIVES A CONSTANT WARNING. (WARNING LIGHT BURNS CONTINUALLY WHILE AIRCRAFT BATTERY OR EXTERNAL POWER SOURCE IS CONNECTED.) (Cont)		
	If disconnecting the sensing elements from the circuit removed the short, reconnect one end of the loop to the circuit and disconnect the elements one at a time until the short is isolated.	Replace faulty element.
Sensing element connector shorted.	Carefully disconnect each sensing element connector (paragraph 6-275) and check for metal chips, dirt, bent pin or broken ceramic insulator.	Clean, reconnect and safety-wire connectors or replace sensing element.
WARNING LIGHT FAILS TO ILLUMINATE WHEN TEST SWITCH IS ENERGIZED.		
Sensing elements broken or disconnected.	Perform continuity test. (Refer to paragraph 6-273.)	Replace defective elements.

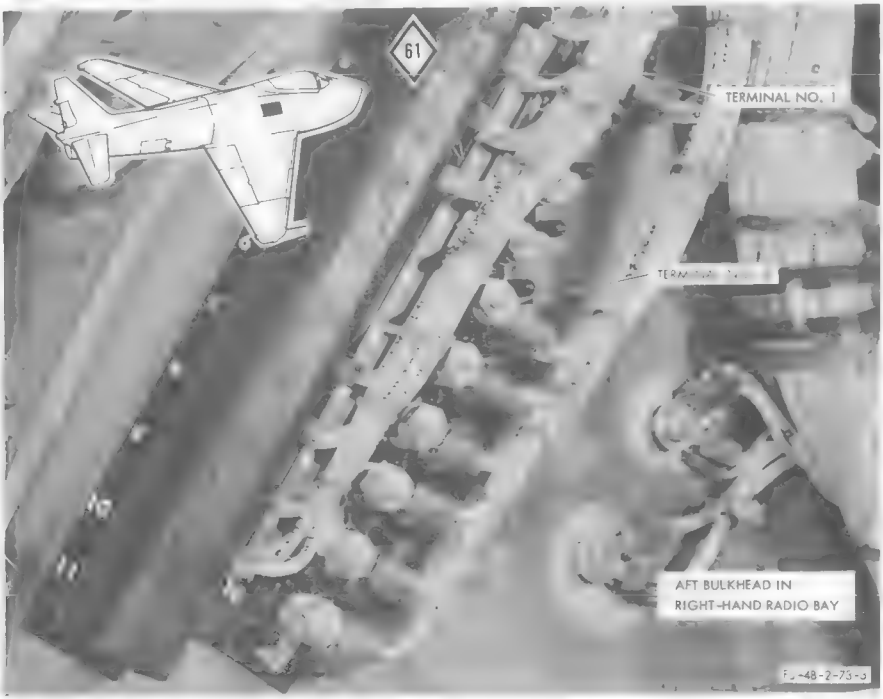


Figure No. 6-58. Engine Fire Detector System Test Points

6-273. TESTING ENGINE FIRE DETECTOR SYSTEM.

6-274. The following equipment is required for testing the engine fire detector system:

A standard type megohmmeter with a capacity of 100 volts dc.
A Wheatstone bridge (Leeds and Northrup 5300, Type S, or equivalent).
A calibrated decade box (Heath Kit Model DR-1, or equivalent).
One 28-volt d-c constant voltage electrical power source.

The engine fire detector system should be checked by testing sensing element circuit insulation, continuity and wire resistance and the control box "trip resistance." To test sensing element circuit insulation, proceed as follows:

Note

The purpose of these tests is to verify that the resistance to ground of each sensing element wire is of the correct value.

a. Ensure that airplane is thermally stable at a temperature of 70°F (±10°F).

b. Test compressor compartment circuit as follows: (1) Remove terminal strip guard and connect one lead of the megger to terminal "1" of terminal strip No. 61 in the right-hand radio bay. (See figure 6-58.) Connect the other lead of the megger to a convenient uninsulated portion of the airplane structure. (2) Using the megger, measure and record the resistance to ground. The minimum allowable resistance is 150 kilohms.

c. Test the burner compartment circuit using the same general procedure as given in step b. (1) With one lead of the megger connected to ground and the other lead connected to terminal "5" of terminal strip No. 61, measure and record the resistance. The minimum allowable resistance is 120 kilohms.

Note

If the resistance for either circuit is less than the values noted in steps b. and c., respectively, test for faulty wiring or a defective sensing element by a megger test of the circuit components. The 809060 sensing element should have a minimum value of 1200 kilohms and the 809120 sensing element should have a minimum value of 600 kilohms.

To test sensing element continuity and wire resistance, proceed as follows:

Note

The purpose of these tests is to confirm that the sensing element wire resistance is within tolerance and that the element wire has proper continuity.

a. Test compressor compartment sensing element as follows: (1) Connect a Wheatstone bridge between terminal "1" of terminal strip No. 61 and ground. (See figure 6-58.) (2) Depress the FIRE DET test switch and measure the resistance in the compressor compartment sensing element. The maximum allowable resistance is 40 ohms.

b. Test burner compartment sensing element using same general procedure as given in step a. (1) Connect a Wheatstone bridge to terminal "5" of terminal strip No. 61 and ground. (See figure 6-58.) (2) Depress the FIRE DET test switch and measure the resistance in the burner compartment sensing element. Maximum allowable resistance is 50 ohms.

To test control box "trip resistance," proceed as follows:

Note

The object of these tests is to establish that the control box operates at the desired resistance setting. The "trip resistance" refers to the value of external sensing element resistance which will trigger the control box and cause the warning light to illuminate. The "trip resistance" and control box internal potentiometer setting will not necessarily be the same value.

a. Examine units for condition and proper installation. Cap which covers range adjustment screw should be safety-wired.

b. Connect a 28-volt d-c external power source to the airplane's power receptacle marked "EXTERNAL 28-V. DC" and allow approximately 2 minutes for control boxes to warm up.

Note

Place the d-c power switch in the "OFF" position before servicing airplane with 28-volt d-c external power.

c. Depress FIRE DET test switch and verify that fire warning lights, located on instrument panel, illuminate.

d. Test compressor compartment control box as follows: (1) Connect a decade box between terminal "1" of terminal strip No. 61 and ground. (2) Turn the resistance to maximum (3000 ohms). (3) Slowly decrease the variable resistance value until the warning light first comes on. Measure this resistance. This value is the "trip resistance" and must be 300 (±15) ohms for the forward circuit.

e. Test burner compartment control box using same general procedure given in step d. "Trip resistance" for burner compartment circuit must be 300 (±15) ohms.

Note

Step f. is to be accomplished in shop only.

f. If control unit resistance value is not within the tolerances listed in steps d. and e., adjust control box setting as follows: (1) Remove safety wire and knurled cap protecting range adjustment screw. (2) Turn adjustment screw full counterclockwise. (3) Set decade box to 300 (±15) ohms. (4) Turn adjustment screw clockwise until operation of light is indicated. Replace cap hand-tight and safety-wire. Repeat step d. or f., depending on which control box is being tested.

g. Remove test equipment and replace terminal guard.

h. Disconnect 28-volt d-c external power source.

6-275. REMOVING AND INSTALLING ENGINE FIRE DETECTOR SYSTEM SENSING ELEMENTS.

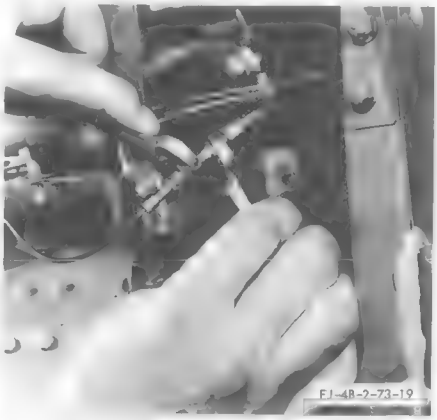
Note These instructions are typical for any of the fire detector system sensing elements.

REMOVING

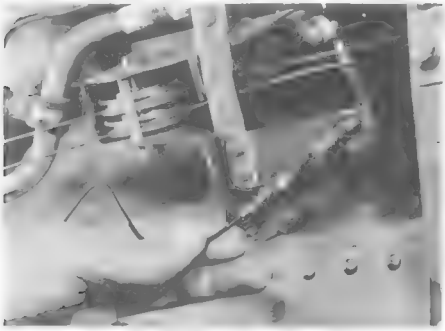
1 Remove clamp and safety wire from connector.

2 Loosen union nuts using two wrenches to prevent twisting and possible damage.

Caution Do not attempt to disconnect sensing element at any point except that shown. Do not use pliers or other tools which might damage connector or element.



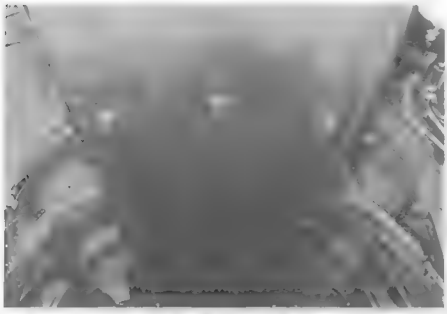
3 Unscrew connector by hand and carefully slide apart.

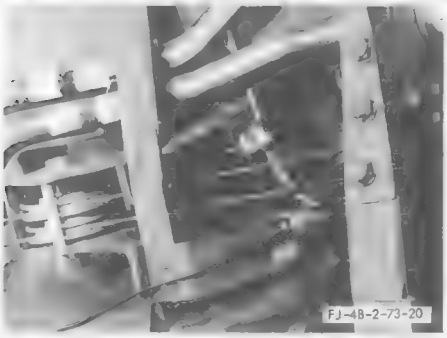


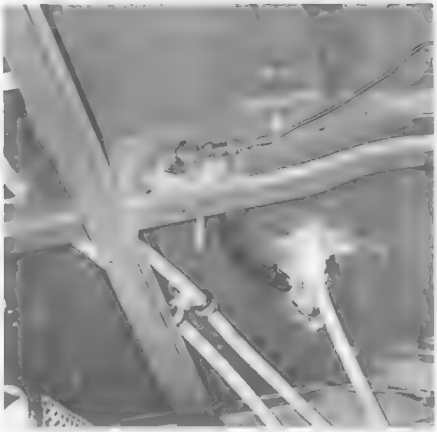
4 Repeat steps 1, 2 and 3 for connector on opposite end of sensing element.

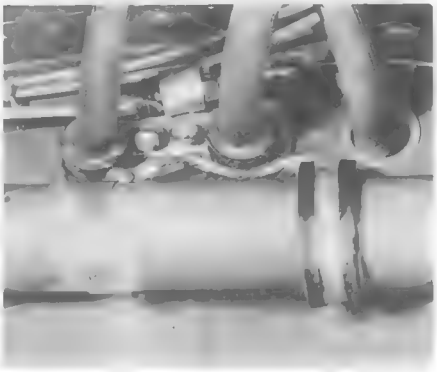
5 Unclamp sensing element and remove.

TYPICAL SENSING ELEMENT AND CONNECTOR INSTALLATIONS









INSTALLING

1 Route sensing element through proper clamps and clamp in position.

2 Slip connectors together and tighten finger-tight. Torque connectors from 20 to 30 inch-pounds.

3 Safety-wire connector and clamp in position.

4 Repeat steps 1, 2 and 3 for connector on other end of sensing element.

5 Perform functional test of engine fire detector system. (Refer to paragraph 6-273.)

FJ-4B-2-73-21

MATERIALS AND TOOLS NEEDED

GLASS TAPE (ITEM 127, MATERIALS LIST)

CEMENT (ITEM 140, MATERIALS LIST)

CRIMPING TOOL, BUCHANAN NO. C-24B, WITH STOP INCORPORATED TO HOLD A DIAMETER OF 0.059 INCH

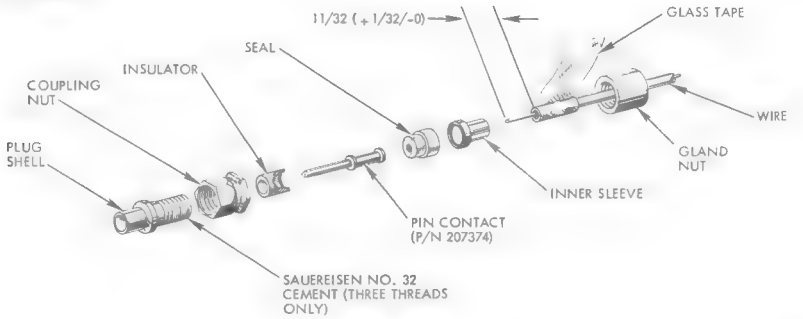
WIRE STRIPPER

TORQUE WRENCH, 30-35 INCH-POUND

COMMON PLIERS WITH JAWS TAPED

Note The crimping tool must have an adjustable cam-type stop, so that the jaws can be set at 0.059 inch in the fully closed position. (Use shank of a No. 53 drill for proper diameter.)

840838 ASSEMBLY



ASSEMBLY OF 840838 ASSEMBLY

1 Cut end of wire square and even. Remove 11/32 inch of insulation. Make sure that no loose strands of insulation remain.

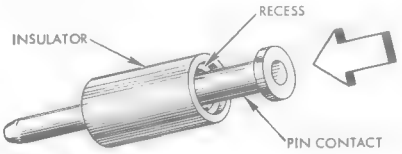
2 Slide gland nut and inner sleeve on wire as shown.

3 Apply several turns of one-half inch glass tape (item 127, materials list) around wire insulation even with end of insulation.

4 Position seal on wire and force it over tape with a twisting motion so that tape fits into cup in seal.

5 Slide insulator over contact, making absolutely sure that recessed end of insulator faces seal.

6 Place contact in position on wire and crimp contact with crimping tool.



7 Place coupling nut over threaded end of plug shell and position plug shell over the insulator, the contact and the seal. Apply thin coat of cement (item 140, materials list) to first three threads of plug shell.

8 Seat inner sleeve against seal and screw gland nut onto plug shell.

9 Connect coupling nut to a dummy receptacle shell and tighten. Hold dummy receptacle shell with torque wrench and torque gland nut from 30 to 35 inch-pounds using pliers with jaws taped.

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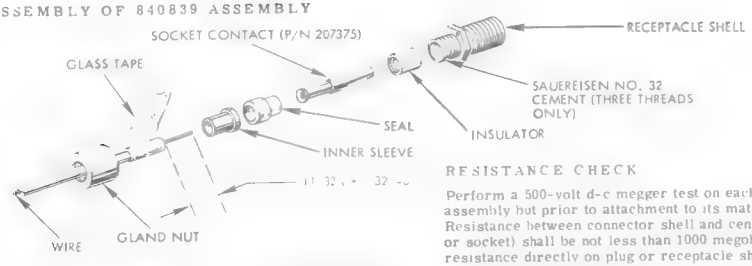
Figure No. 6-58A. Assembly of Engine Fire Detector System Connectors (Sheet 1)

ASSEMBLY OF 840839 ASSEMBLY



Perform steps 1 through 6 of previous (840838) assembly.

7 Position receptacle shell over the insulator, the contact and the seal. Apply a thin coat of cement (item 140, materials list) to first three threads of receptacle shell.

8 Seat inner sleeve against seal and screw gland nut onto receptacle shell.

9 Hold hex head of receptacle shell with torque wrench and torque gland nut from 30 to 35 inch-pounds using pliers with jaws taped.

RESISTANCE CHECK

Perform a 500-volt d-c megger test on each connector after assembly but prior to attachment to its mating assembly. Resistance between connector shell and center contact (pin or socket) shall be not less than 1000 megohms. Measure resistance directly on plug or receptacle shell; do not measure through coupling nut or gland nut.

COMPLETED ASSEMBLY (UNITS 840839 AND 840838)

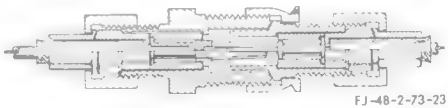


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Figure No. 6-58A. Assembly of Engine Fire Detector System Connectors (Sheet 2)

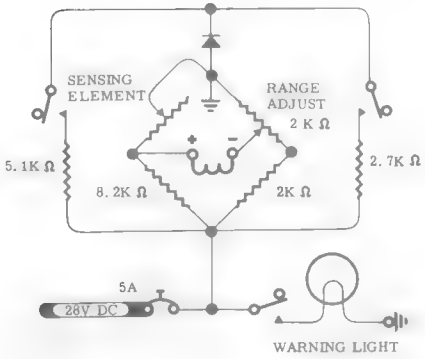

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Figure No. 6-58B. Engine Fire Detector System—Elementary Schematic

TACHOMETER SYSTEM

6-276. TACHOMETER SYSTEM.

6-277. The tachometer indicating system (figure 6-59) provides an indication of the speed of rotation of the engine. The speed is read as a percent of the rated rpm of the engine. A tachometer generator, installed on the engine, generates three-phase alternating current which drives a synchronous motor in the indicator located on the instrument panel.

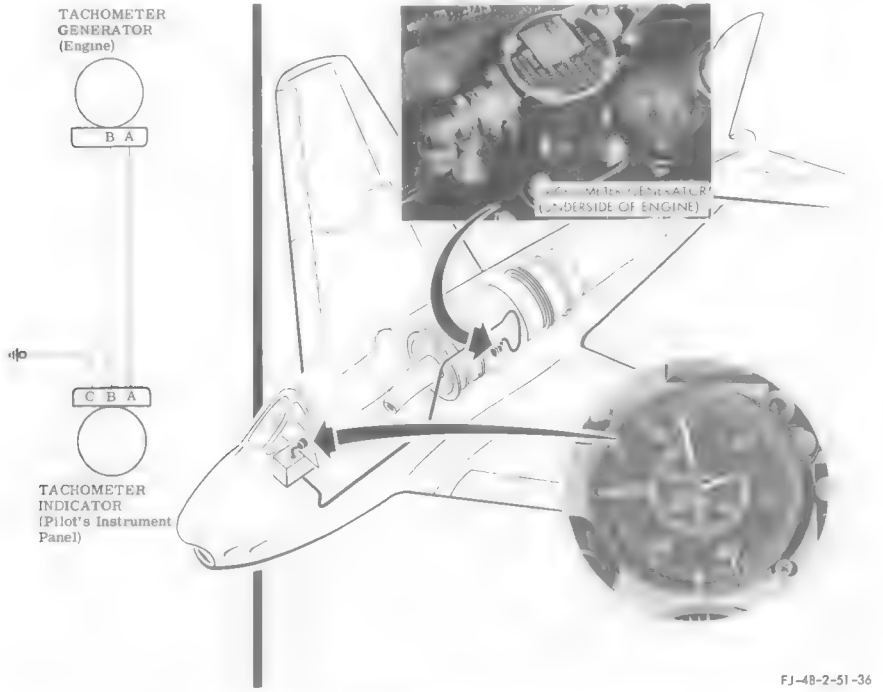


Figure No. 6-59. Tachometer Indicating System

6-278. TROUBLE SHOOTING TACHOMETER SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
LOW OR INCORRECT INDICATION.		
Friction in tachometer indicator or defective indicator.	Substitute a test tachometer indicator and perform a functional check of tachometer indicating system. (Refer to paragraph 6-279.)	If system checks out satisfactorily, replace indicator. (Refer to paragraphs 6-7 and 6-8.)

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
LOW OR INCORRECT INDICATION. (Cont)		
Defective tachometer generator.		Replace tachometer generator. (Refer to paragraph 6-284.)
NO INDICATION.		
Open or shorted lead; wiring improperly connected or defective.	Check for proper connections. Make continuity check of circuit.	Correct opens or shorts; replace defective wiring.
Defective tachometer indicator.		Replace indicator. (Refer to paragraphs 6-7 and 6-8.)
Broken tachometer generator drive shaft.	Remove generator and visually examine shaft.	Replace generator. (Refer to paragraph 6-284.)
Tachometer generator rotor weak or demagnetized.	Perform functional check of tachometer indicating system. (Refer to paragraph 6-279.)	Replace generator. (Refer to paragraph 6-284.)
INTERMITTENT INDICATION.		
Poor connection or short circuit in leads; broken leads.	Visually examine circuit for proper connections; make continuity check of circuit.	Remake connections; repair or replace defective wiring.
Defective tachometer indicator.		Replace indicator. (Refer to paragraphs 6-7 and 6-8.)
Defective tachometer generator.		Replace generator. (Refer to paragraph 6-284.)
INDICATING POINTER OSCILLATES.		
Defective tachometer indicator.		Replace indicator. (Refer to paragraphs 6-7 and 6-8.)
ENGINE WARMING UP, INDICATOR NOT FOLLOWING.		
Broken leads, wiring improperly connected or dirty connections.	Visually examine all connections to be sure they are clean and tight.	Replace defective wiring.
Defective tachometer generator.	Perform continuity and resistance check of tachometer generator. (Refer to paragraph 6-283.) Remove generator and examine for broken drive shaft.	Replace defective generator. (Refer to paragraph 6-284.)
Defective tachometer indicator.		Replace indicator. (Refer to paragraphs 6-7 and 6-8.)
INDICATOR READS BACKWARDS.		
Wires crossed and lead connections reversed at the indicator or generator.	Visually check all wiring and connections.	Make necessary corrections to circuit.

6-279. FUNCTIONAL CHECK OF TACHOMETER SYSTEM.

6-280. The functional operation and accuracy of the tachometer indicating system is checked with the Jetcal Analyzer (B & H Instrument Co., Inc., Model BH-112J) and additional B & H tachometer check equipment. The Jetcal check circuit and the airplane's tachometer are paralleled so that comparable readings can be made simultaneously to determine the accuracy of the system within ±0.1% rpm. The tachometer check instrument panel on the Jetcal Analyzer (figure 6-60) has a COARSE scale (calibrated from 0 to 102% in 1% increments) for use when engine is being run up and a FINE scale (calibrated from 95 to 102% in 0.1% increments) to be used after the engine has reached 95% rpm. The FINE scale affords extreme accuracy in checking the higher speed range of the engine. The tachometer indicating system should be checked each time the system has been disconnected, following engine replacement, or after replacing the tachometer generator or tachometer indicator. To perform a check of the tachometer indicating system, proceed as follows:

a. Place selector switch SW-1 in "OFF" position (figure 6-49); connect power inlet cable to Jetcal at receptacle P-1 and to 110-volt, 50- to 400-cycle, a-c power.

b. Connect tachometer check adapter to instrument cable (figure 6-49) and instrument cable to Jetcal at receptacle P-3.

c. Loosen tension screw at lower right side of tachometer indicator on instrument panel and remove indicator from panel.

d. Disconnect electrical plug from rear of tachometer indicator.

e. Connect tachometer check adapter (two receptacles) to airplane's tachometer cable and to tachometer indicator.

f. Place switch SW-7 in "MECH. ZERO" position.

g. Place selector switch SW-1 in "TACH" position.

h. Adjust MECH. ZERO knob of GALVO-2 until GALVO-2 indicates zero.

i. Place switch SW-7 in "COARSE" position.

j. Start engine (paragraph 1-9) and run at idle rpm. Tachometer indicator should read the correct idle rpm percent for station elevation. (See figure 5-9.)

k. Adjust PERCENT R.P.M. dial knob until GALVO-2 reads zero.

l. Read percent rpm on scale marked "COARSE."

m. Difference between scale reading and tachometer indication is the error of the tachometer indicating system and should not exceed ±1.25 percent.

n. Advance power control lever until engine stabilizes at normal operating power.

o. Place switch SW-7 in "FINE" position.

p. Adjust PERCENT R.P.M. dial knob until GALVO-2 reads zero.

q. Read percent rpm on scale marked "FINE."

r. Difference between scale reading and tachometer indication is the error of the tachometer indicating system and should not exceed ±1.25 percent.

s. Advance power control lever to full power.

t. Adjust PERCENT R.P.M. dial knob until GALVO-2 reads zero.

u. Read percent rpm on scale marked "FINE."

v. Difference between scale reading and tachometer indication is the error of the tachometer indicating system and should not exceed ±1.0 percent.

NOTE

If any of the tolerances are exceeded, substitute a test tachometer indicator in the airplane's circuit and repeat check. If system checks out satisfactorily, replace the tachometer indicator. If system is not satisfactory, trouble shoot the tachometer indicating system and the engine. (Refer to paragraphs 5-3 and 6-278.)

w. Shut down engine.

x. Place switch SW-7 in "MECH. ZERO" position and place selector switch SW-1 in "OFF" position.

y. Remove tachometer check adapter and cables.

z. Reconnect airplane tachometer cable to the rear of the tachometer indicator; position indicator in the instrument panel and secure by tightening tension screw.

6-281. TACHOMETER GENERATOR.

6-282. The tachometer generator, installed on the lower aft face of the engine accessory gear box, is a two-pole, electric generator. The permanent magnet rotor revolves with the tachometer drive shaft which is geared directly to the engine. The rotor revolves at 4200 rpm for the maximum rated speed of the engine (approximately 8300 rpm) and produces a maximum voltage of 20 to 21 volts. As the rotor revolves, it induces three-phase alternating current in the stator. This three-phase alternating current is transmitted to the indicator producing a rotating magnetic field in the indicator stator.

6-283. CHECKING TACHOMETER GENERATOR RESISTANCE AND CONTINUITY. The resistance and continuity of the tachometer generator may be checked without removing the generator from the engine. To check the tachometer generator, proceed as follows:

a. Remove right-hand wheel well access door.

b. Disconnect plug from generator.

c. Using an ohmmeter, check resistance between respective sets of connector pins or between pins and frame of generator. Resistance values should be within one ohm of each other, and there should be no evidence of discontinuity.

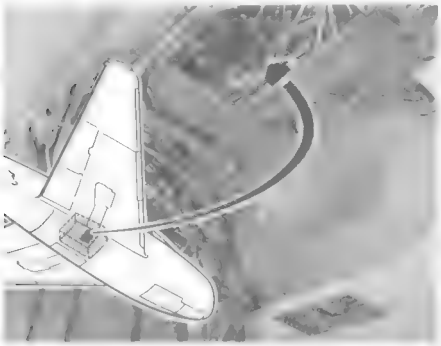
d. Reconnect plug and safety-wire connector.

e. Replace access door.

6-284. REMOVING AND INSTALLING TACHOMETER GENERATOR.

REMOVING

1 Remove right-hand wheel well access door.



2 Remove four nuts securing tachometer generator to mounting pad.

3 Remove tachometer generator from mounting pad on the engine.

4 Cut safety wire and remove electrical connector. Tie plug in bag and cover the receptacle with masking tape.

5 Remove tachometer generator from the airplane.

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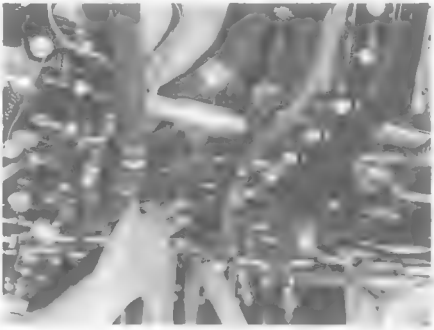
INSTALLING

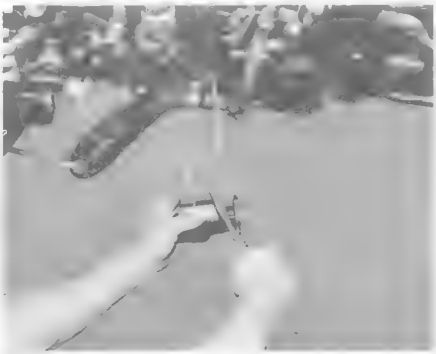
1 Gain access through the right-hand wheel well access door and locate electrical connector for tachometer generator.

2 Make electrical connection to the generator and safety-wire plug with AN995F32 wire.



Note For ease of installation, the electrical connection and safety wiring should be accomplished before installing tachometer generator.

3 Fit base of tachometer generator over retaining bolts and seat unit on mounting pad.



4 Secure with four mounting nuts.

5 Perform an operational check of tachometer indicating system. (Refer to paragraph 6-279.)

6 Install access door.

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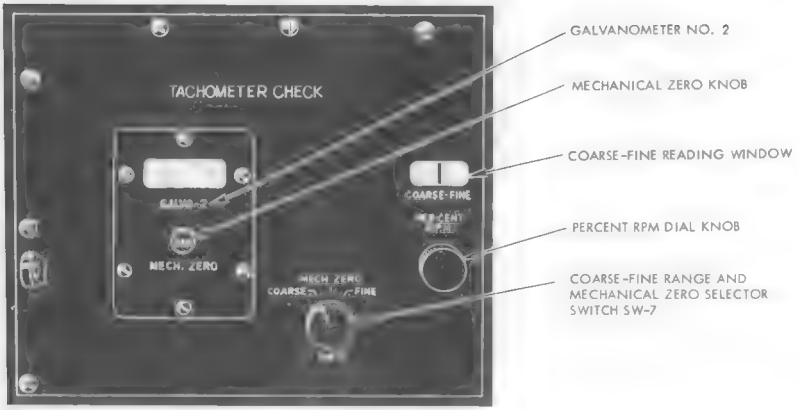

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Figure No. 6-60. Jetcal Tachometer Check Instrument Panel

6-285. TACHOMETER INDICATOR.

6-286. The tachometer indicator contains a synchronous motor and a magnetic drag indicating mechanism which provides indications of the actual speed of rotation of the engine. When the stator of the indicator receives three-phase alternating current generated at the generator, the rotating magnetic field causes the four-pole, dual rotor assembly to revolve. On the indicator dial, two concentrically mounted pointers indicate in percent rpm with reference to two separate scales. The short pointer rotates through 270 degrees on the inner scale to indicate from 0 to 50% rpm. The long pointer remains stationary at the 50% rpm graduation until the short pointer is superimposed on the long pointer. Both pointers then rotate approximately 340 degrees on the outer scale to indicate from 50 through 110% rpm. The tachometer indicates 100% when the tachometer generator is driven at 4200 rpm.

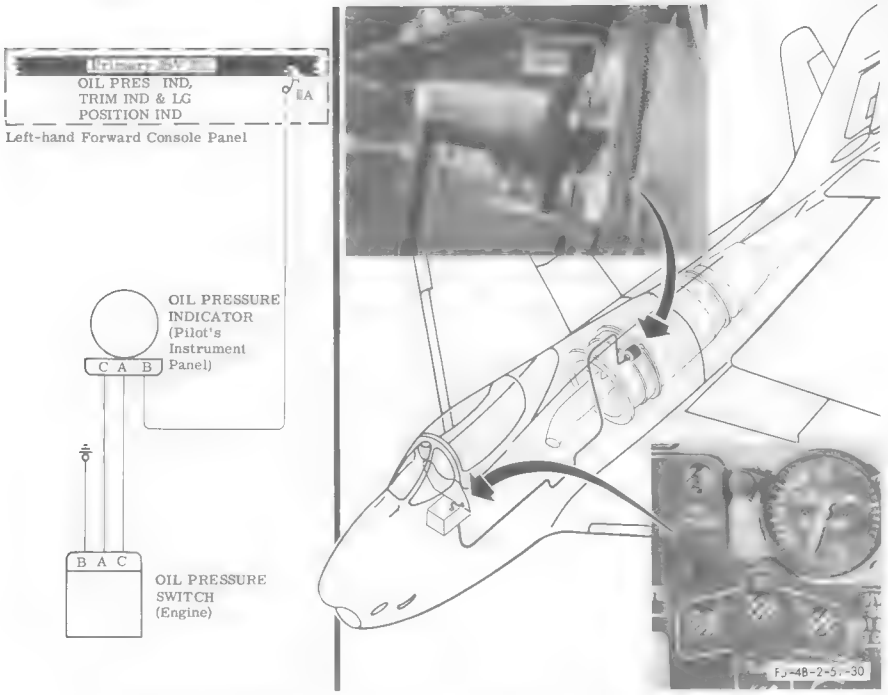


Figure No. 6-61. Oil Pressure Indicating System—Airplanes 139531i through 143542k

OIL PRESSURE INDICATING SYSTEM

6-287. OIL PRESSURE INDICATING SYSTEM— AIRPLANES 139531i THROUGH 143542k.

6-288. The oil pressure indicating system (figure 6-61) provides an accurate indication of the operating pressure of the engine's oil. The system is composed of an oil pressure switch installed on the engine, a three-position electromagnetic oil pressure indicator, located on the pilot's instrument panel, and associated wiring. Power for the system is received from the primary 28-volt d-c bus. The system is electrically protected by a 5-ampere circuit breaker (OIL PRESS IND, TRIM IND & LG POSITION IND) located on the left-hand forward console.

6-289. TROUBLE SHOOTING OIL PRESSURE INDICATING SYSTEM— AIRPLANES 139531i THROUGH 143542k.

TEST EQUIPMENT: D-C voltmeter.
Ohmmeter.

SYSTEM CONDITIONS: External power on airplane.
OIL PRESS IND, TRIM IND & LG POSITION IND circuit breaker pushed in.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
"LO" INDICATION AT ALL TIMES (OIL PRESSURE KNOWN TO BE CORRECT).			
Defective indicator.	Check between test point EPD and ground.	28 volts dc.	Replace indicator.
		Zero volts.	Continue trouble shooting.
No power to indicator.	Check between test point GBD and ground.	28 volts dc.	Replace defective wire to test point EPD.
		Zero volts.	Replace defective power wire or circuit breaker. If no power is availabe at test point PDB, refer to paragraph 8-60, Trouble Shooting D-C Power Distribution System.
Defective wiring between indicator and oil pressure switch.	Check for continuity between test points EPE and EPB and between EPF and EPC.	No continuity.	Repair or replace defective wire.
		Continuity.	Continue trouble shooting.
Defective oil pressure switch.	Refer to paragraph 6-298.	None.	Adjust or replace switch as required.
"HI" INDICATION (OIL PRESSURE KNOWN TO BE CORRECT). ENGINE MUST BE OPERATING FOR THIS CHECK.			
Defective indicator.	Check between test point EPE and ground.	28 volts dc.	Replace defective indicator.
		Zero volts.	Check for crossed wires and reconnect as necessary. If wires are not crossed, adjust or replace oil pressure switch. (Refer to paragraph 6-298.)

Figure No. 6-62. Oil Pressure Indications— Airplanes 139531i through 143542k

6-290. (DELETED.)

6-291. (Deleted.)

6-292. (Deleted.)

6-293. OIL PRESSURE INDICATOR—AIRPLANES 139531i THROUGH 143542k.

6-294. The oil pressure indicator is a solenoid-operated, three-position indicator which gives two positive indications and a neutral or power-off indication. The neutral position is determined by spring tension and appears as "LO" on a barber-pole background in the center position of the indicator. This indication represents low oil pressure or indicates no operating power is being supplied to the indicator. The two positive indications are obtained by energizing either of two solenoids which overcome the spring tension. They appear as "N" on a plain background, representing normal operating pressure, and "HI" on a barber-pole background. (See figure 6-62.)

6-295. OIL PRESSURE SWITCH—AIRPLANES 139531i THROUGH 143542k.

6-296. The electromagnetic action within the indicator is initiated by a pressure-actuated switch installed on the upper left side of the engine's front main bearing support. The switch contains two single-pole, double-throw microswitches with one side of each switch at ground potential. The switch may complete either the normal pressure circuit or the high-pressure circuit to the indicator, depending upon oil pressure conditions. The switch is adjusted to actuate on increasing pressure at 25 (±1) psi to complete the normal pressure circuit and to actuate at 38 (+2/—0) psi to complete the high-pressure circuit. On decreasing pressure, the switch is adjusted to open the high-pressure circuit before 40 (+0/—2) psi and to open the normal pressure circuit before 25 (±1) psi. When the oil pressure is less than normal, neither microswitch will be closed. The oil pressure switch is accessible through the left-hand engine access door.

6-297. OPERATIONAL CHECK OF OIL PRESSURE INDICATING SYSTEM — AIRPLANES 139531i THROUGH 143542k.

Note

Two men are required to perform this check.

a. Remove left wheel well access door.

b. Disconnect engine oil pressure line to oil pressure switch and install a "T" fitting between this flexible line and the oil pressure switch.

c. Connect an auxiliary engine oil pressure gage (calibrated in psi) to "T" fitting.

d. Open left-hand engine access door.

e. Check electrical plug at oil pressure switch for tightness.

f. Start engine. Oil pressure indicator should read "LO" before engine is started.

g. On increasing pressure, auxiliary engine oil pressure gage should read from 0 through 25 psi as the engine warms up while the indicator still indicates "LO." At approximately 25 (±1) psi gage reading, the oil pressure indicator should change from "LO" to "N."

h. Slowly accelerate engine to 75% rpm.

i. Auxiliary oil pressure gage should increase from approximately 25 to approximately 40 psi; the oil pressure indicator should remain at "N."

j. Slowly accelerate engine to 100% rpm.

k. If engine oil pressure should exceed 38 (+2/—0) psi, oil pressure indicator should change from "N" to "HI," indicating a malfunction in the engine oil system. Refer to paragraph 5-3 for trouble shooting engine oil system.

l. After engine oil system trouble has been corrected and at 40 (+0/—2) psi on decreasing pressure auxiliary gage reading, the oil pressure indicator should change from "HI" to "N."

m. Slowly decelerate the engine. While auxiliary oil pressure gage indicates from approximately 40 to 25 psi, oil pressure indicator should indicate "N."

n. Continue to decelerate engine. At 25 (±1) psi, on decreasing pressure, the oil pressure indicator should change from "N" to "LO."

o. Shut down engine. Oil pressure indicator should remain at "LO."

p. If oil pressure indicator does not indicate correctly, replace oil pressure switch with a switch that has been checked for accurate switch settings. (Refer to paragraph 6-299.)

q. Remove auxiliary oil pressure gage and "T" fitting. Reconnect engine oil pressure line to oil pressure switch.

r. Close left-hand engine access door.

s. Replace left wheel well access door.

6-298. SWITCH REPLACEMENT CHECK—AIRPLANES 139531i THROUGH 143542k.

a. Using a regulator valve and master pressure gage, connect the pressure switch (No. SW-2E, U. S. Gauge Co.) to a source of pressure.

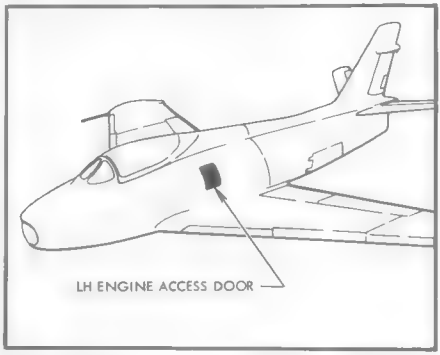
b. With no air pressure applied, make continuity check between test points EP1 and EP2 and test points EP1 and EP3. Continuity check should show open.

c. Slowly increase pressure to switch.

d. At 25 (±1) psi, repeat continuity check. Continuity check should show open between test points EP1 and EP3 and closed between test points EP1 and EP2.

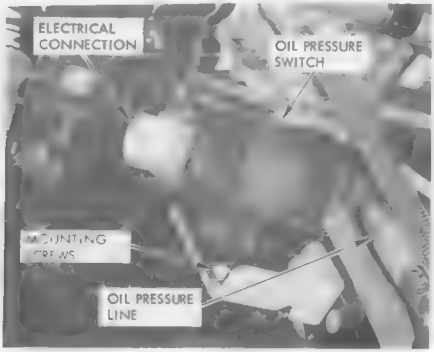
e. At 38 (+2/—0) psi, repeat continuity check. Continuity check should show open between test points EP1 and EP2 and closed between test points EP1 and EP3.

6-299. REMOVING AND INSTALLING OIL PRESSURE SWITCH—AIRPLANES 139531i THROUGH 143542k.



REMOVING

1. Remove left-hand engine access door.
2. Cut safety wire and remove electrical connector.
3. Remove oil pressure line from port on aft end of switch. Cap port and line.



4. Remove two mounting screws on each side of switch mounting bracket and remove switch.

INSTALLING

1. Reverse above steps for correct installation procedure.
2. Perform operational check of oil pressure indicating system. (Refer to paragraph 6-290.)

FJ-48-2-51-32

6-300. OIL PRESSURE INDICATING SYSTEM—AIRPLANES 1435431 AND SUBSEQUENT.

6-301. The oil pressure indicating system (figure 6-63) provides a direct reading of engine oil pressure. The system is composed of an oil pressure transmitter, installed on the engine, a continuous reading oil pressure indicator, installed on the instrument panel, and associated wiring. Power for the system is supplied from the single-phase, 26-volt a-c bus. The system is electrically protected by a one-ampere fuse (HYD & OIL PRESS IND) located on the right-hand rear vertical console.

6-302. TROUBLE SHOOTING OIL PRESSURE INDICATING SYSTEM—AIRPLANES 1435431 AND SUBSEQUENT.

TEST EQUIPMENT: Ohmmeter.
A-C voltmeter.

SYSTEM CONDITIONS: External power on airplane.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
NO INDICATION OR INCORRECT INDICATION.			
Defective power input.	Check between test points EPL, EPH and ground.	26 volts ac.	Continue trouble shooting.
		Zero volts at either test point.	Replace defective wiring between test point EPL or EPH and XVB.
Defective interconnecting wiring.	Check between test points EPN and EPJ.	Zero ohms.	Continue trouble shooting.
		Other than zero ohms.	Replace defective wire segment between test points EPN and EPJ.
	Check between test points EPM and EPK.	Zero ohms.	Continue trouble shooting.
		Other than zero ohms.	Replace defective wire segment between test points EPM and EPK.
Defective oil pressure transmitter.	Check transmitter using master autosyn indicator (Bendix Part No. 13647-1-A or equivalent).	None.	Adjust or replace defective transmitter.
Defective oil pressure indicator.	Check indicator using master autosyn transmitter (Bendix Part No. 13695-1-A or equivalent).	None.	Replace defective indicator.

6-303. (Deleted.)

6-304. OPERATIONAL CHECK OF OIL PRESSURE INDICATING SYSTEM—AIRPLANES 1435431 AND SUBSEQUENT.

6-305. An operational check of the oil pressure indicating system should be accomplished following replacement of the engine, the oil pressure indicator, the oil pressure transmitter or associated wiring.

Note

Two men are required to perform this check.

a. Remove left wheel well access door.

b. Disconnect oil pressure sensing line at the oil pump and install a "T" fitting between this line and the oil pump.

c. Connect an auxiliary engine oil pressure gage, calibrated in psi, to the "T" fitting.

d. Start engine and compare auxiliary engine oil pressure gage reading with engine oil pressure indicator reading. If engine oil pressure indicator reading is incorrect, trouble shoot the oil pressure indicating system. (Refer to paragraph 6-302.)

6-306. OIL PRESSURE INDICATOR—AIRPLANES 1435431 AND SUBSEQUENT.

6-307. The oil pressure indicator is a continuous reading synchro-type instrument. The dial is calibrated from 0 to 70 psi in increments of 10 psi.

6-308. OIL PRESSURE TRANSMITTER—AIRPLANES 1435431 AND SUBSEQUENT.

6-309. The oil pressure transmitter is mounted on the left side of the front main bearing support housing on the engine at approximately the 10 o'clock position. The autosyn-type transmitter functions to convert oil pressure into electrical energy which is transmitted to the oil pressure indicator through appropriate wiring.

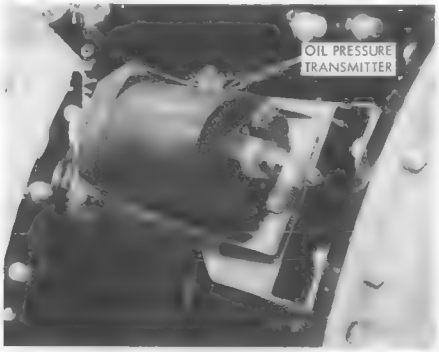
6-310. REMOVING AND INSTALLING OIL PRESSURE TRANSMITTER—AIRPLANES 1435431 AND SUBSEQUENT.

REMOVING

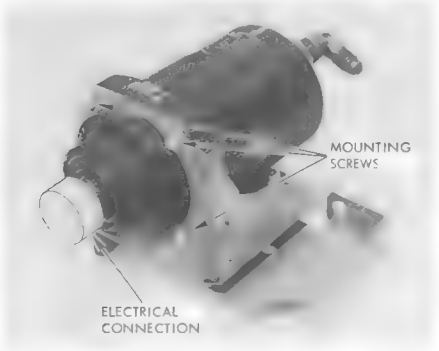
Note Access through left-hand engine access door.

1 Cut safety wire and remove electrical connection.

2 Remove oil pressure line. Cap line and port.



3 Remove four mounting screws.



4 Remove transmitter.

5 Remove elbow, jam nut and union. Cap all openings.

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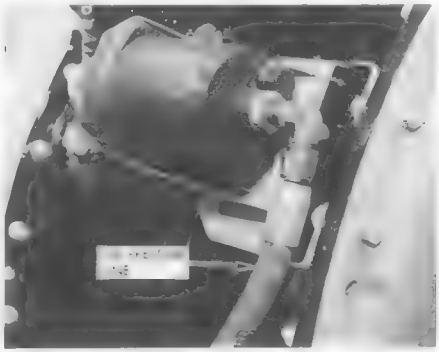
INSTALLING

1 Install union in vent port and elbow in pressure port.

2 Position transmitter in mounting bracket.

3 Install four mounting screws; safety-wire screws.

4 Connect oil pressure line.



5 Connect electrical connection. Safety-wire electrical connection.

Note For check-out of system, refer to paragraph 6-304.

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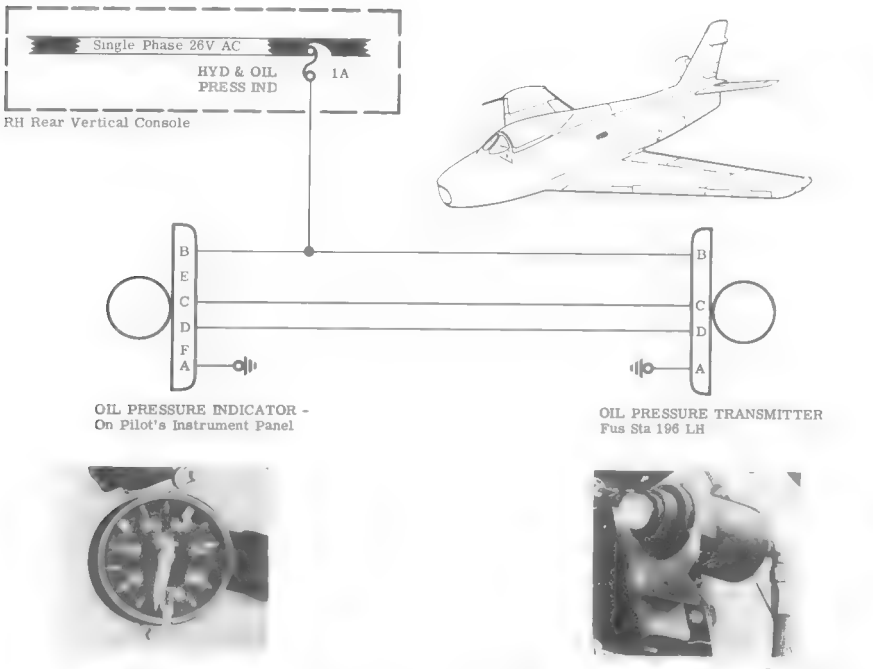


Figure No. 6-63. Oil Pressure Indicating System—Airplanes 143543I and Subsequent

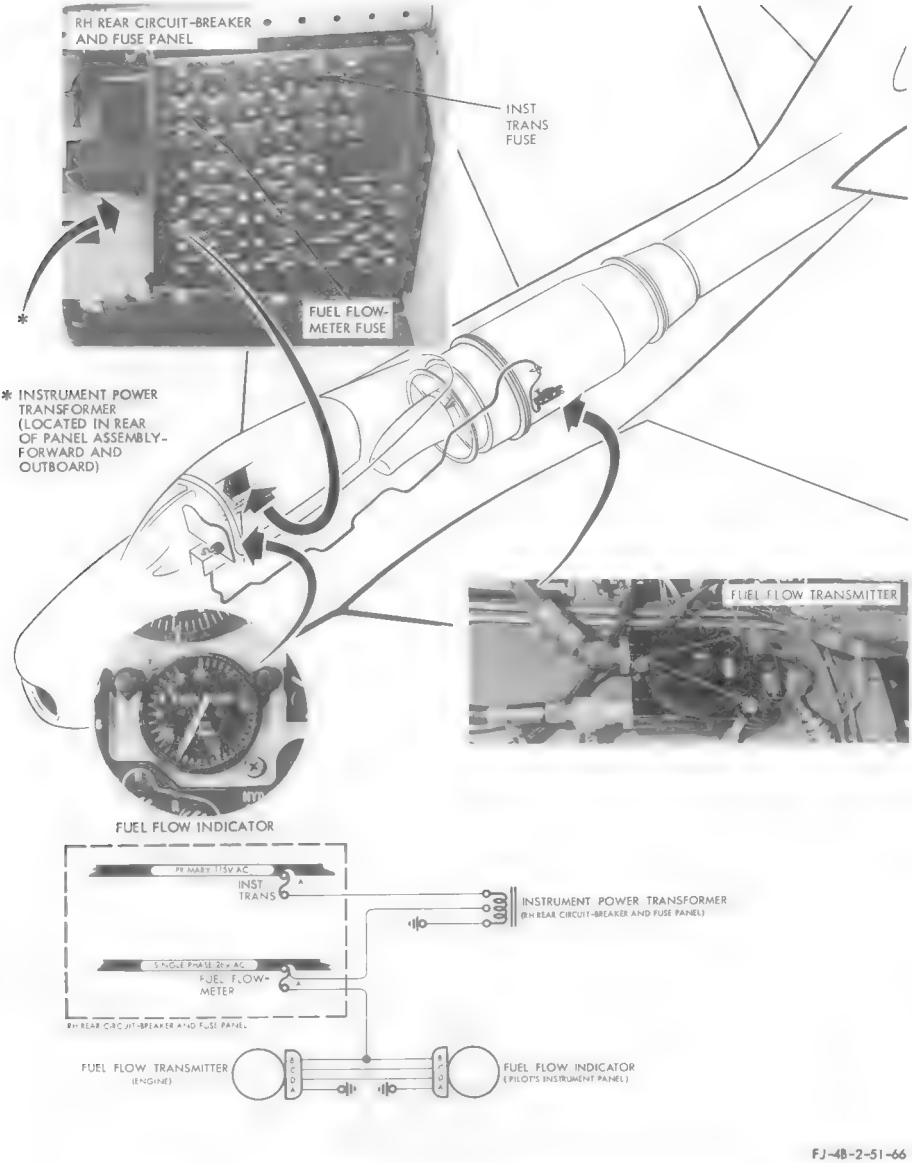
FUEL FLOW INDICATING SYSTEM

6-311. FUEL FLOW INDICATING SYSTEM.

6-312. The fuel flow indicating system provides a continuous indication of the rate that fuel is being delivered to the engine in pounds per hour. The indicating system consists of a synchro-type transmitter which senses the fuel flow and transmits the information in the form of electrical energy through interconnecting wiring to the synchro-type receiver indicator. The system is powered by 26-volt, 400-cycle alternating current from the 26-volt, single-phase a-c bus. A one-ampere fuse (FUEL FLOW METER) protects the circuit from shorts and is located on the right-hand rear vertical console.

6-313. TROUBLE SHOOTING FUEL FLOW INDICATING SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
INDICATOR POINTER SLUGGISH BUT TRIES TO FOLLOW TRANSMITTER.		
No power on one rotor.	Check power connection to both indicator and transmitter.	Connect hot lead to pin "B."
Defective indicator.	Perform a fuel flow indicator test. (Refer to paragraph 6-320.)	Replace defective indicator (R88-I-1206-100).
POINTER SWINGS BACK AND FORTH IN LIMITED ARC ACROSS TOP OF DIAL.		
Open transmitter ground lead.	Check pin "A" on the transmitter.	Connect wiring to pin "A."
Reversed power leads.	Check all wiring.	Reverse connections if necessary.
POINTER SWINGS BACK AND FORTH IN LIMITED ARC ACROSS BOTTOM OF DIAL.		
Open indicator ground lead.	Check pin "A" on the indicator.	Connect wiring to pin "A."
POINTER SWINGS AT SIDE OF DIAL—NO SQUEAL HEARD INSIDE INDICATOR.		
Open stator leads.	Check pins "C" and "D" on the transmitter and on the indicator.	Make connections as necessary.
POINTER SWINGS AT SIDE OF DIAL—SQUEAL HEARD INSIDE INDICATOR.		
Short circuit between power lead and stator lead.	Check for possible short between pins "B" and "C," or between pins "B" and "D."	Eliminate short.
Reversed connection between power lead and stator lead.	Check for reversals in connections between pins "B" and "C" and between pins "B" and "D" on both the transmitter and the indicator.	Reverse connections as necessary.
POINTER MAKES COMPLETE ROTATION IN REVERSE DIRECTION.		
Reversed stator leads.	Check connections for reversal at pins "C" and "D."	Reverse connections as necessary.
LOW INDICATION.		
Defective indicator.	Perform a fuel flow indicator test. (Refer to paragraph 6-320.)	Replace indicator (R88-I-1206-100).



FJ-4B-2-51-66

Figure No. 6-64. Fuel Flow Indicating System and Instrument Transformer

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
SLIGHT OR NO MOVEMENT OF POINTER.		
Short circuit between stator leads.	Check for shorts between pins "C" and "D."	Eliminate short.
Short circuit between power leads.	Check for short in lead from power source to pin "B" on both transmitter and indicator.	Eliminate short.
Defective indicator.	Perform a fuel flow indicator test. (Refer to paragraph 6-320.)	Replace indicator (R88-I-1206-100).
Clogged or dirty fuel pressure lines or connection to transmitter.	Remove fuel pressure line to fuel flow transmitter. Examine end of line and fitting for dirt.	Remove any foreign material.
POINTER SPINS (MOTORIZED).		
Power lead reversals, power and stator lead short circuit or reversals or intermittent contact.	Check all connections. Check for continuity and shorts. Eliminate shorts.	Check wiring and replace indicator (R88-I-1206-100). (Refer to paragraphs 6-7 and 6-8.)
Defective indicator.	Perform a fuel flow indicator test. (Refer to paragraph 6-320.)	Replace indicator (R88-I-1206-100). (Refer to paragraphs 6-7 and 6-8.)

6-314. OPERATIONAL CHECK OF FUEL FLOW INDICATING SYSTEM.

6-315. To perform an operational check of the fuel flow indicating system, proceed as follows:

a. Place d-c power switch, located on the right-hand forward console, in "OFF" position.

b. Connect external power to the airplane.

c. Place INST. AC POWER switch, located on the right-hand forward console, in the "NO. 1 INV." position.

d. Fuel flow indicator should read below "0" pounds per hour.

e. Remove external power from airplane.

f. Start engine and run at "IDLE." (See figure 1-15.)

g. Check that indicator pointer moves smoothly over dial as engine warms up. With engine running at "IDLE," fuel flow should read 950 to 1050 pounds per hour. If indicator operation is faulty, replace indicator. (Refer to paragraphs 6-7 and 6-8.) If trouble persists, replace fuel flow transmitter. (Refer to paragraph 6-324.)

6-316. TESTING FUEL FLOW INDICATING SYSTEM.

6-317. The fuel flow indicating system may be tested by using an autosyn instrument field tester and an appropriate tester adapter cable. To test the system, proceed as follows:

CAUTION

Always remove power before making a change or substitution in testing the system.

a. Remove left-hand wheel well access door.

b. Disconnect electrical plug from fuel flow transmitter connector.

c. Connect autosyn instrument field tester to the airplane wiring with appropriate tester adapter cable.

d. Push in button on face of autosyn tester and rotate the button until the pointer is set at "0" (12 o'clock position).

e. Be sure d-c power switch, located on the right-hand forward console, is in "OFF" position.

f. Connect a source of external power to airplane.

g. Check to see that instrument power off warning light (INST PWR OFF), located on the right-hand forward console panel, does not illuminate. The light should be off to indicate that a-c power is available to the instrument system.

h. Rotate autosyn tester pointer to "180" position.

i. Observe fuel flow indicator on the pilot's instrument panel. Indicator pointer should be in same position as autosyn tester pointer.

j. If indicator pointer agrees with autosyn tester pointer, the indicator and the airplane's wiring can be eliminated as sources of system trouble.

k. If indicator pointer does not agree with the autosyn tester pointer, indicator should be tested individually (paragraph 6-320) and the airplane wiring should be checked for continuity.

l. Remove source of external power.

m. Remove autosyn field tester and equipment.

n. Reconnect electrical plug to fuel flow transmitter receptacle.

o. Replace wheel well access door.

p. Perform an operational check of system. (Refer to paragraph 6-314.)

6-318. FUEL FLOW INDICATOR.

6-319. The fuel flow indicator, located on the lower left-hand corner of the instrument panel, indicates the rate of flow which has been determined at the transmitter. The indicator consists of an autosyn, a graduated dial and a pointer assembly which is mechanically linked to the autosyn. The scale ranges from 0 to 12 and, when any reading is multiplied by 1000, the indication is the actual rate of fuel flow in pounds per hour.

6-320. TESTING FUEL FLOW INDICATOR. The fuel flow indicator may be tested individually by using the autosyn instrument field tester and a test transmitter. To test the indicator, proceed as follows:

a. Remove fuel flow indicator by loosening tension screw at lower right corner of indicator and pulling it from panel. Remove electrical connector from rear of indicator.

b. Connect indicator in autosyn test circuit with an autosyn precision transmitter (Type 13695-1-A, Eclipse-Pioneer, or equivalent).

c. At the following test points, check the indicator reading (and pointer movement in degrees) against controlled fuel flow through test transmitter.

PRECISION TEST TRANSMITTER	FUEL FLOW INDICATOR		TOLERANCES			
			SCALE ERROR		FRICTION ERROR	
POUNDS PER HOUR	POUNDS PER HOUR	DEGREES POINTER MOVEMENT	POUNDS PER HOUR (±)	DEGREES POINTER MOVEMENT (±)	POUNDS PER HOUR (±)	DEGREES POINTER MOVEMENT (±)
1,000	1,000	95	50	1.4	100	2.8
2,000	2,000	176	50	1.4	100	2.8
3,000	3,000	256	50	1.4	100	2.8
6,000	6,000	286	100	1.4	200	2.8
8,000	8,000	307	100	1.4	200	2.8
12,000	12,000	346	100	1.4	200	2.8

d. If tolerances are exceeded, replace fuel flow indicator; otherwise, reinstall indicator in instrument panel. Connect electrical connector to receptacle on rear of case, position indicator in clamp installed in cutout on panel and tighten tension screw at lower right side of face of indicator.

e. Perform an operational check of fuel flow indicating system. (Refer to paragraph 6-314.)

6-321. FUEL FLOW TRANSMITTER.

6-322. The fuel flow transmitter is mounted on the lower left-hand side of the engine compressor housing. The transmitter consists of a mechanical device for measuring the rate of fuel flow and a transmitting autosyn. The measuring device consists of a vane assembly, a calibrated spring and a bar magnet, all of which are mounted on a common shaft. A ring-type magnet, separated from the bar magnet by the autosyn mounting frame, is attached to the autosyn rotor shaft. As fuel enters the metering chamber and displaces the vane, rotation is transmitted to the autosyn rotor shaft by means of the magnetic coupling. An integral relief valve protects the fuel flow transmitter from excessive fuel pressure.

6-323. TESTING FUEL FLOW TRANSMITTER. The fuel flow transmitter may be tested individually by using the autosyn instrument field tester and a test indicator. To test the transmitter, proceed as follows:

a. Remove fuel flow transmitter. (Refer to paragraph 6-324.)

b. Install transmitter in a test assembly so that fuel flow through it may be checked against a standard.

c. Connect an autosyn servo test fuel flow indicator (Type 13647-1-A, Eclipse-Pioneer), or an indicator whose accuracy has been determined to be ±1.5 circular degrees or better, to the transmitter in the test circuit.

d. At the following test points, check the indicator reading (and pointer movement in degrees) against the controlled fuel flow through the transmitter:

PRECISION TEST TRANSMITTER	FUEL FLOW INDICATOR		TOLERANCES			
			SCALE ERROR		FRICTION ERROR	
POUNDS PER HOUR	POUNDS PER HOUR	DEGREES POINTER MOVEMENT	POUNDS PER HOUR (±)	DEGREES POINTER MOVEMENT (±)	POUNDS PER HOUR (±)	DEGREES POINTER MOVEMENT (±)
1,000	1,000	95	50	1.4	20	0.7
2,000	2,000	176	75	2.1	20	0.7
3,000	3,000	256	100	2.8	20	0.7
6,000	6,000	286	200	5.6	40	0.7
8,000	8,000	307	400	11.2	40	0.7
12,000	12,000	346	800	22.4	40	0.7

e. If indicator readings exceed tolerances, fuel flow transmitter should be checked for leakage. Perform steps f. through j.

f. Remove fuel equipment from transmitter.

g. Remove electrical connector from transmitter.

h. While fuel chamber at transmitter is still wet with fuel, but with all free fluid drained out, connect a source of clean, dry air to transmitter pressure connection.

i. Apply a pressure of 40 psi.

Note

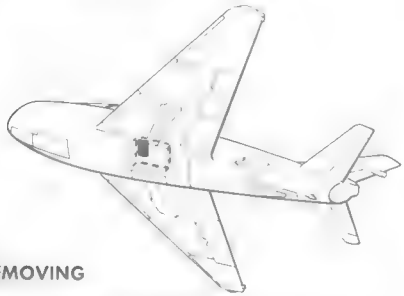
Measure the pressure with a gage capable of detecting a change of 1.8 psi. Allow the pressure to remain on for a sufficient length of time to allow the air and transmitter temperatures to become equalized.

j. There should be no perceptible change in air pressure after the supply has been shut off for 5 minutes.

k. Reinstall or replace fuel flow transmitter. (Refer to paragraph 6-324.)

l. Perform an operational check of system. (Refer to paragraph 6-314.)

6-324. REMOVING AND INSTALLING FUEL FLOW TRANSMITTER.



REMOVING

Note Extreme care should be taken to prevent dirt from entering system.

1. Remove left-hand wheel well access door.

2. Cut safety wire and remove electrical connectors; tie transmitter pigtail plug in protective bag and cover airplane receptacle with masking tape.

3. Remove three fuel lines from fittings on transmitter and cover all open lines and ports with masking tape.

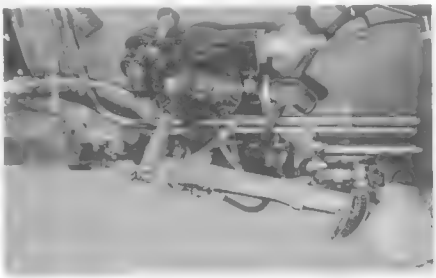
4. Cut safety wire and remove nuts from three retaining bolts.

5. Lift transmitter from retaining bolts and remove from airplane.

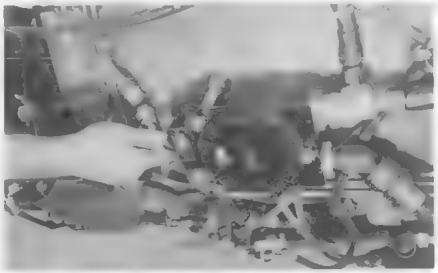
FJ-4B-2-51-70

INSTALLING

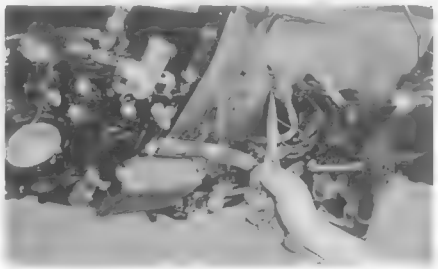
1. Position transmitter over three retaining bolts and secure with nuts. Torque nuts from 80 to 85 inch-pounds.



2. Remove tape from fuel lines and transmitter ports and make standard connection of lines to ports.

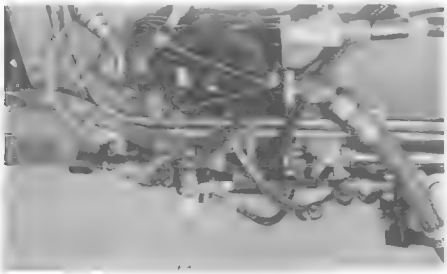


3. Remove protective bag from electrical plug and remove tape from receptacle; make electrical connection and safety-wire with AN995F32 wire.



FJ-4B-2-51-69

4 Safety-wire three retaining bolts with AN995F41 wire.



5 Perform an operational check of fuel flow indicating system. (Refer to paragraph 6-314.)

6 Install the left-hand wheel well access door.

FJ-4B-2-51-191

FUEL QUANTITY INDICATING SYSTEM

6-325. FUEL QUANTITY INDICATING SYSTEM.

6-326. The fuel quantity indicating system is a capacitance-type system, designed to indicate, in pounds, the amount of fuel available to the engine at any time. The system consists of an indicator power unit, 16 tank units, a reference condenser, a selector switch, a selector relay and associated wiring.

6-327. FUNCTION OF FUEL QUANTITY INDICATING SYSTEM.

6-328. The fuel quantity indicating system functions on the principle that the capacitance of a condenser is determined by the dielectric constant of the insulating medium between the condenser plates. The dielectric within the tanks is either a liquid fuel, a mixture of air and vapor or both. The indicator circuit is a continuously rebalanced bridge circuit in which the capacitance of the tank units is compared with that of a reference condenser. The difference in capacitance of the bridge circuit is amplified by the indicator amplifier which operates a null seeking motor attached to a potentiometer arm of the bridge circuit. As the fuel quantity changes, the motor repositions the potentiometer arm. The indicator pointer is mechanically attached to the potentiometer arm and indicates the fuel quantity in pounds on the indicator face. For schematic diagram of fuel quantity indicating system, see figure 6-65.

6-329. TROUBLE SHOOTING FUEL QUANTITY INDICATING SYSTEM.

Note

Isolation of defective circuitry or components, or improper adjustments of the fuel quantity indicating system, may involve a complete system check as described in paragraph 6-330. However, certain types of malfunctions may be resolved by the following trouble shooting chart.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
NO FUEL QUANTITY INDICATION.		
Defective indicator-power unit.	Disconnect coaxial fittings from indicator and substitute an MD-1 (R88-T-0941-005) tester for the tank units. (Refer to paragraph 6-330.)	Replace indicator-power unit if the indicator pointer does not move with change in capacity.
INDICATOR POINTER ROTATES CONTINUOUSLY.		
Defective wiring, tank unit or indicator-power unit.	Disconnect coaxial fittings from indicator and substitute an MD-1 tester for tank units and wiring. (Refer to paragraph 6-330.) If indicator pointer performs properly according to steps m. through o. in paragraph 6-331, an open circuit or short circuited component, or wiring, is indicated.	Locate and remove open circuit or short circuited component, or wiring, and replace.
INDICATOR POINTER OSCILLATES WITH LEVEL FLIGHT.		
Defective indicator-power unit or wiring.	Disconnect coaxial fittings from indicator and substitute an MD-1 tester for tank units and wiring. (Refer to paragraph 6-330.)	
	If oscillation continues, the indicator-power unit is defective.	Replace indicator-power unit.
	If oscillation stops, defective wiring or tank unit is indicated.	Trace intermittent short or open circuit in coaxial leads and repair.

TEST EQUIPMENT: A-C voltmeter.
D-C voltmeter.
Ohmmeter.

SYSTEM CONDITIONS: INST VIBRATOR circuit breaker engaged.
External power on airplane.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
INDICATOR POINTER READS A CONSTANT VALUE.			
Defective indicator-power unit, a-c power supply or shorted HI Z lead.	Check between test point EQA and ground.	115 volts ac.	Replace indicator-power unit.
		Zero volts.	Perform wire segment continuity check between test points EQA and XCE and replace defective wire segment or fuse.
	Check for shorted HI Z lead.	None.	Repair or replace as required.
INDICATOR READS SUMP FUEL REGARDLESS OF SELECTOR SWITCH POSITION.			
	Note FUEL GAGE READS selector switch positioned to "TOTAL" for these checks.		
Defective selector switch, selector relay or wire segment.	Check between test point EQC and ground.	28 volts dc.	Replace defective selector relay.
		Zero volts.	Continue trouble shooting.
	Check between test point EQB and ground.	28 volts dc.	Perform wire segment continuity check between test points EQB and EQC and replace defective wire segment or selector switch.
		Zero volts.	Perform wire segment continuity check between test points EQB and PGE and replace defective wire segment or circuit breaker.
SYSTEM BLOWS FUSES.			
Defective indicator-power unit or wiring.	Disconnect center electrical connector from indicator; then, replace fuse.	None.	
	If fuse does not blow, the indicator-power unit is defective.		Replace indicator.
	If fuse continues to blow, the airplane's wiring to pin "B" is shorted to ground or to another pin.		Repair or replace wiring as necessary.
INDICATOR OFF CALIBRATION.			
Defective tank unit.	Isolate defective tank unit. (Refer to paragraph 6-330.)	None.	Replace defective unit.

6-330. CHECKING AND ADJUSTING FUEL QUANTITY INDICATING SYSTEM.

6-331. To check and adjust fuel quantity indicating system, proceed as follows:

Note

Fuel tanks must be empty for these checks. If necessary to defuel, refer to paragraph 1-35.

a. Loosen indicator by loosening clamp screw on instrument panel at lower right-hand side of indicator.

b. Slide indicator aft from instrument panel and disconnect leads marked 209-51051-103, 209-51051-105 and 209-51051-155 from rear of indicator.

c. Connect MD-2A tester (Specification MIL-T-4687) "A" cable to open 209-51051-103 lead and tester "B" cable to open 209-51051-105 lead through suitable patch cables. (See figure 6-68.) Ground the center conductor of the open 209-51051-155 lead.

Note

A-C power to operate the MD-2A tester may be obtained from the No. 2 inverter through suitable connections if external a-c power is not readily available. These connections may be made by removing the terminal cover of terminal strip 101 which is located in the right-hand side of the nose equipment bay, and by connecting a suitable adapter to either terminal No. 8 or 14. The inverter may be energized by applying 28-volt d-c external power to the airplane. Do not connect into the No. 1 inverter system since the additional 40 watts consumed by the tester would overload the No. 1 inverter system.

d. Push in INST VIBRATOR circuit breaker, located on left-hand radio bay circuit-breaker panel, and place FUEL GAGE READS selector switch in "TOTAL" position.

e. Turn on the MD-2A tester and allow 30 seconds for warm-up. Select "RESISTANCE—MEGOHMS" and check the insulation resistance of "A" to ground, "B" to ground and "A" to "B." Minimum values are as follows:

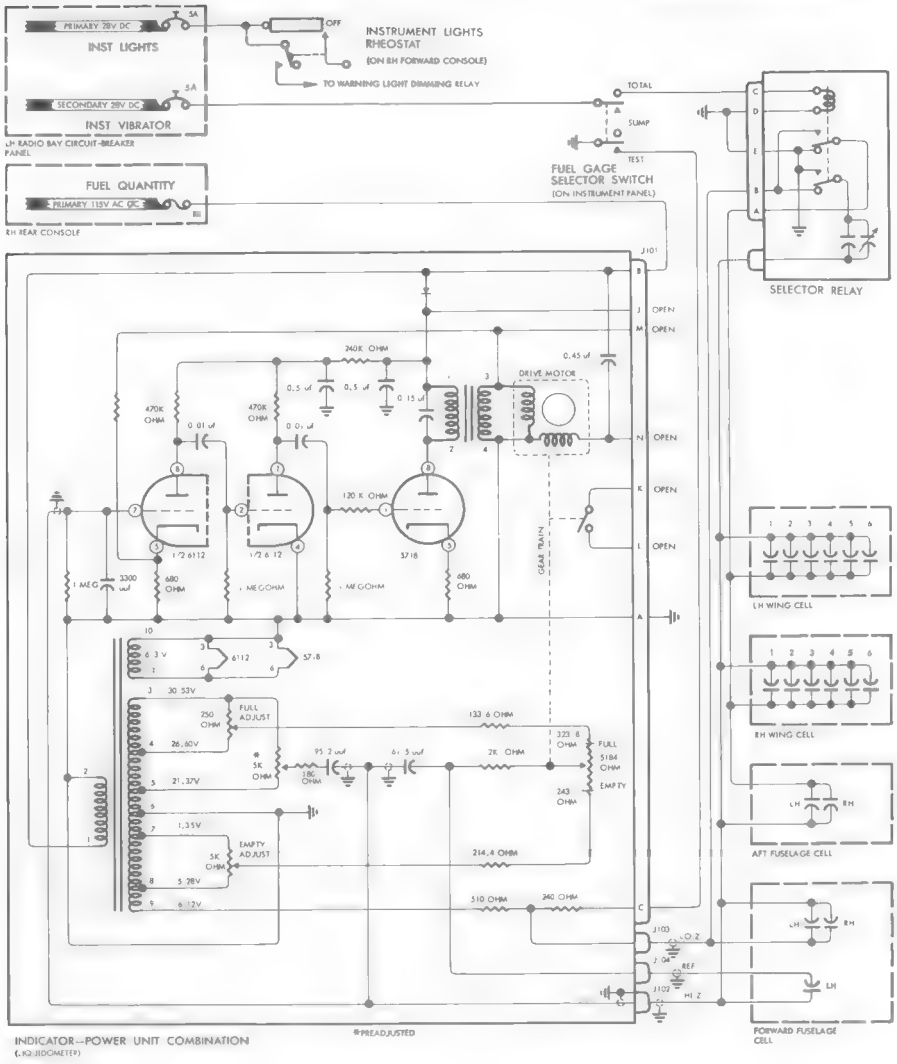


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Figure No. 6-65. Fuel Quantity Indicating System

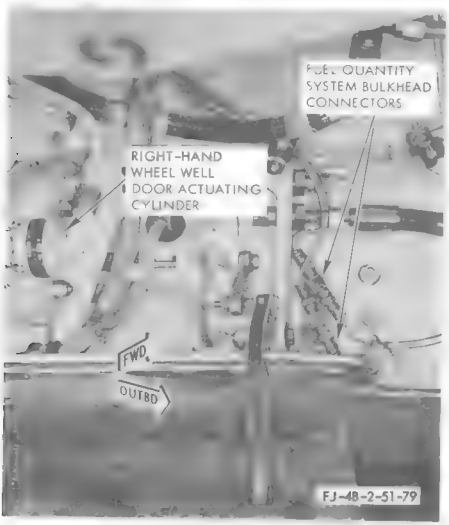


Figure No. 6-66. Right-hand Wheel Well Fuel Quantity Indicating System Coaxial Fittings

"A" to ground—20 megohms
"B" to ground—20 megohms
"B" to "C"—40 megohms

f. Select "SUMP" position on FUEL GAGE READS selector switch and repeat step e.

g. Check reference capacitor by disconnecting 209-51051-105 lead from tester "B" cable and connecting "B" cable to 209-51051-155 lead. (See figure 6-68.) Ground center conductor of 209-51051-105 lead. Repeat step e.

h. Connect MD-2A tester as in step c. and select "CAPACITANCE—MMF" on tester. Select "TOTAL" on FUEL GAGE READS selector switch. Disconnect 209-51051-139, 209-51051-111, 209-51051-141 and 209-51051-113 leads from spar fittings in right-hand wheel well. (See figure 6-66.)

i. Read capacitance on tester. This is the empty capacitance of the fuselage probes. Value should be between 236.0 and 251.2 mmf, with a nominal value of 243.6 mmf. Check for open or short circuit in wiring if capacitance value deviates greatly from this nominal value. For a more complete list of capacitance values, refer to paragraph 6-332.

j. Connect MD-2A tester as in step g. and read value. This is the capacitance of the reference capacitor and should read 25.5 (±0.5) mmf. If value deviates greatly, check for open or short circuited wiring.

k. Connect MD-2A tester "A" cable to 209-51051-139 lead and "B" cable to 209-51051-141 lead in right-hand wheel well. Value indicated is the capacitance of the left-hand wing tank units and should be between 77.6 and 84.6 mmf, with a nominal value of 81.1 mmf. If value deviates greatly, check for open or short circuited wiring.

l. Connect MD-2A tester "A" and "B" cables to receptacles in right-hand wheel well. Value indicated is for right-hand wing tank units and should coincide with that shown in step k. If value deviates greatly, check for open or short circuited wiring.

m. Disconnect MD-2A tester and reconnect all system leads for system operation.

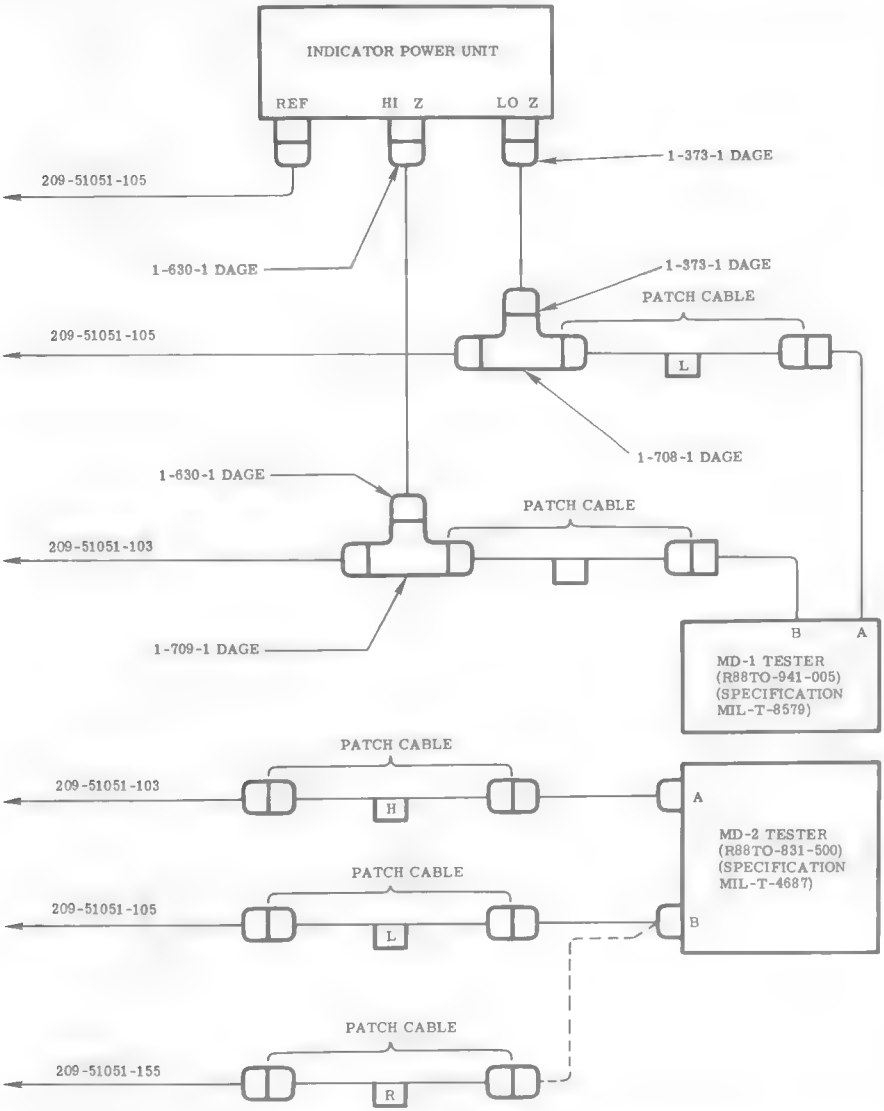
n. Turn system on by operating inverter and engaging INST VIBRATOR circuit breaker.

o. Gain access to "EMPTY" adjustment on back of indicator (figure 6-67) and adjust until indicator reads "0" with FUEL GAGE READS selector switch in "TOTAL" position. Depress FUEL GAGE READS selector switch to "CHECK" position and release. Pointer should move counterclockwise and return to "0." Readjust "0" if necessary. Turn off system power.

p. Disconnect 209-51051-103 and 209-51051-105 leads from rear of indicator and, by means of suitable "T" fittings and test adapters, as shown in figure 6-68, connect a type MD-1 tester (Specification MIL-T-8579) to the indicator. Reconnect 209-51051-103 and 209-51051-105 leads to the "T" fittings.

q. Set MD-1 tester to 375.8 mmf and adjust "FULL" adjustment until the pointer reads "6000." Momentarily depress FUEL GAGE READS selector switch to "CHECK" position to momentarily move pointer off full position; readjust "FULL" adjustment if necessary.

Figure No. 6-67. Fuel Quantity Indicator Empty and Full Adjustments



FJ-48-2-51-81

Figure No. 6-68. Adapters and Cables Used for Checking and Adjusting Fuel Quantity Indicating System (Sheet 1)

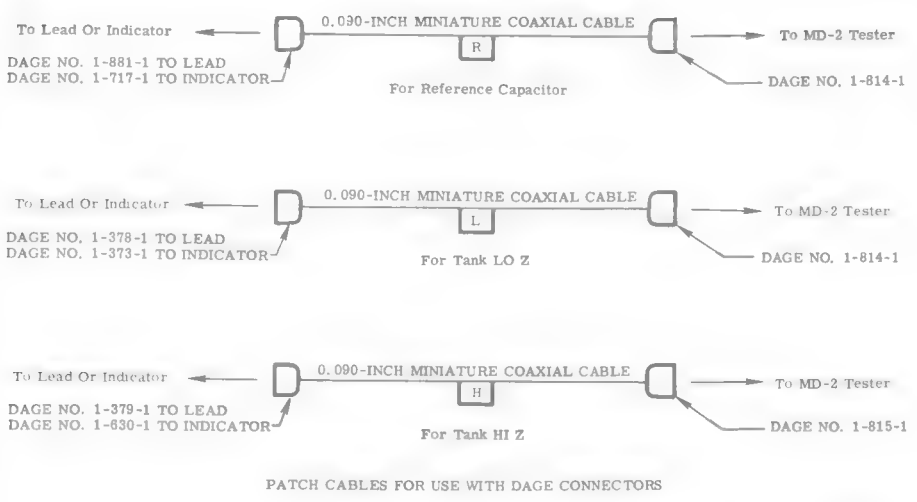


Figure No. 6-68. Adapters and Cables Used for Checking and Adjusting Fuel Quantity Indicating System (Sheet 2)

r. Remove MD-1 tester, "T" fittings and test adapters; reconnect 209-51051-103 and 209-51051-105 leads to the indicator.

s. Check "EMPTY" setting as in steps m. through o. If "EMPTY" setting has changed, repeat steps m. through r.

t. Select "SUMP" position on the FUEL GAGE READS selector switch and adjust capacity trimmer in selector relay until the pointer coincides with "0" on the indicator. The selector relay is located in the electrical equipment compartment in the left-hand fuselage just under the aft canopy deck.

6-332. TANK UNIT CAPACITANCE VALUES.

TANK UNIT LOCATION	PART NUMBER	MD-2A TESTER (A = HI Z)	(B = LO Z)	GROUND CENTER COND.	CAPACITANCE (MICROMICROFARADS) MIN.	NOMINAL	MAX.
Right-hand forward fuselage	EA 804FML-1473	HI Z	LO Z		81.9	82.7	83.5
Left-hand forward fuselage	EA 805FMM-1472	HI Z	LO Z	REF.	82.3	83.1	83.9
		HI Z	REF.	LO Z	24.8	25.3	25.8
Right-hand aft fuselage	EA 804FMK-1474				38.4	38.9	39.4
Left-hand aft fuselage	EA 804FMK-1474	HI Z	LO Z		38.4	38.9	39.4
No. 1 left- and right-hand wing	EA 992GPM-1476	HI Z	LO Z		25.1	25.6	26.1
No. 2 left- and right-hand wing	EA 992GPL-1477	HI Z	LO Z		14.1	14.6	15.1
No. 3 left- and right-hand wing	EA 992GPL-1478	HI Z	LO Z		7.1	7.6	8.1

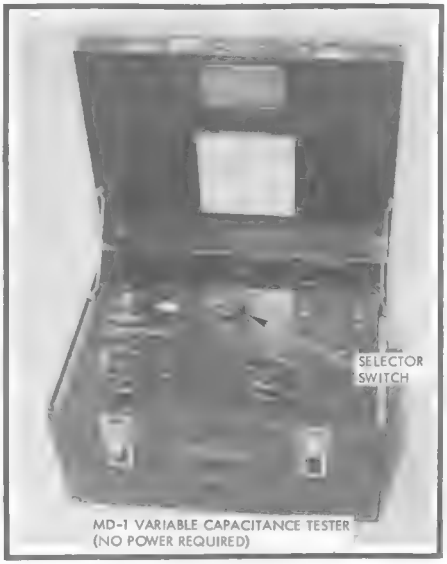
TANK UNIT LOCATION	PART NUMBER	MD-2A TESTER (A — HI Z)	(B — LO Z)	GROUND CENTER COND.	CAPACITANCE (MICROMICROFARADS) MIN.	NOMINAL	MAX.
No. 4 left- and right-hand wing	EA 992GPK-1479	HI Z	LO Z		6.0	6.5	7.0
No. 5 left- and right-hand wing	EA 992GPL-1480	HI Z	LO Z		7.8	8.3	8.8
No. 6 left- and right-hand wing	EA 992GPK-1481	HI Z	LO Z		18.0	18.5	19.0

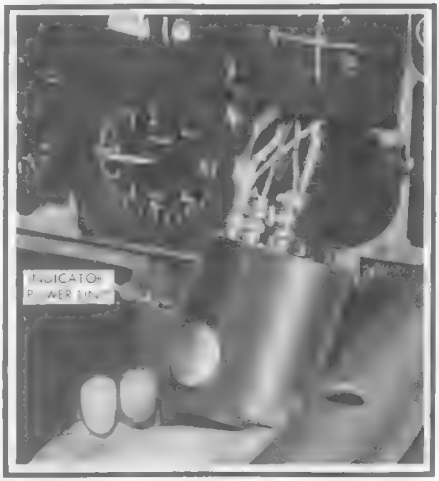
Note

Add 0.5 mmf to maximum readings and subtract 0.5 mmf from minimum readings to compensate for MD-2A instrument error.

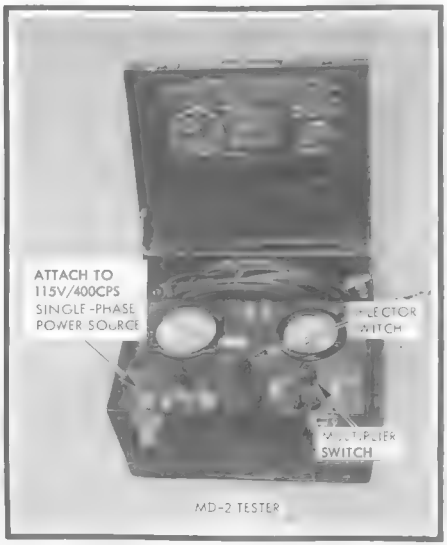
6-333. FUEL QUANTITY INDICATING SYSTEM TANK UNITS.

6-334. All fuel cell tank units incorporate sensing capacitor elements for the fuel quantity indicating system. The sensing elements have a fixed electrode area and spacing and are equipped with receptacles to receive the coaxial cable wiring to the indicator. The connectors are moistureproofed, miniature single contact bayonet base type, with matched bayonet base pins to prevent mismating. (Refer to paragraph 6-335.) The tank units used in the forward and aft fuselage cells compensate for fuel cell contour by being nonlinear in capacity characteristics with respect to the portion of the fuel cell being measured. This non-linearity is necessary to provide linear weight measurements, especially where the lower portions of the fuel cells follow the fuselage contour. The left-hand forward fuselage tank unit contains the reference capacitor which relates the actual volume reading to the fixed weight reading used on the indicator.





The MD-1 tester (R88TO-94-005) is used to substitute capacitance values of tank units and reference condenser and to check the "empty" and "full" adjustments of the indicator power unit. Capacitances required may be selected by rotating the knobs on the front of the tester and observing the readings on the dials.



The MD-2 tester (R88TO-831-500) is used to check calibration accuracy of the tank units by checking the electrostatic capacitance and direct-current resistance.

The tester is designed primarily for field use as a trouble shooting instrument, but it may also be used for shop purposes.

The tester can measure automatically, electrostatic capacitance from 0 to 5000 micromicrofarads.

The tester can measure insulation resistance from 0.5 to 10,000 megohms.

The purpose of the resistance checking section is to enable checks to be made on the resistance existing between the tank unit electrodes as well as the conductors of the associated interconnecting cables with respect to each other and to ground.

Note The 115-volt, 400-cycle power for the tester may be obtained from the airplane's main inverter.

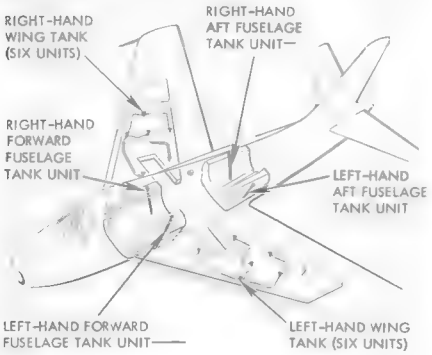
Warning Do not connect MD-2 tester to flight instrument system. The additional 40 watts would overload system inverter.

FJ-48-2-51-83

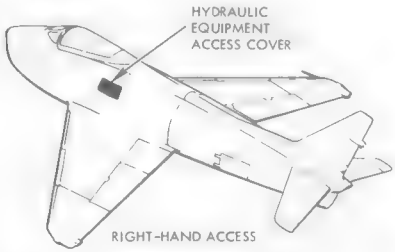
Figure No. 6-69. MD-1 and MD-2A Testers

6-335. REMOVING AND INSTALLING FUEL QUANTITY INDICATING SYSTEM TANK UNITS.

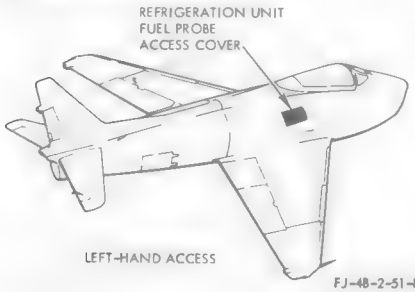
REMOVING



1 Remove access cover (left-hand marked "REFRIGERATION UNIT-FUEL PROBE," right-hand marked "HYDRAULIC EQUIPMENT").

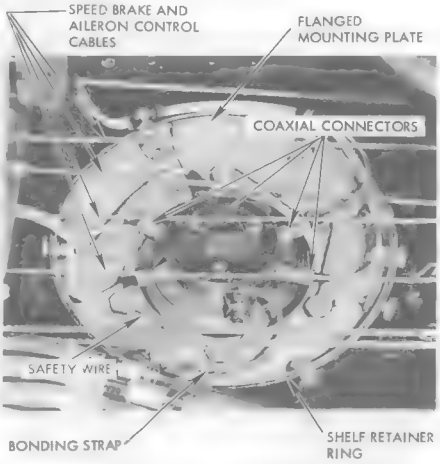


Note Instructions cover left-hand tank unit. Right-hand tank unit is similar.

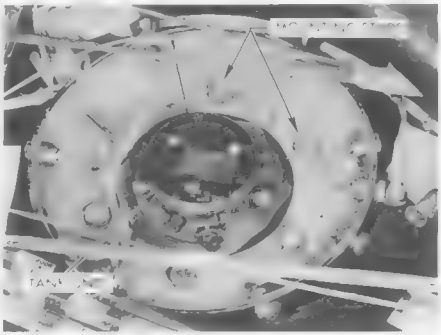


FJ-48-2-51-84

2 Disconnect two aileron control cables and two speed brake control cables at field break to allow them to be pushed aside during tank unit removal.



3 Remove tank unit electrical connectors and bonding strap.



Caution The electrical disconnects must not be handled roughly; tools should be used only as necessary to assist removal.

4 Remove bolts which secure flanged mounting plate to shelf retaining ring.

FJ-48-2-51-85

5 Remove safety wire from studs which hold flanged mounting plate, tank unit and fuel cell together. Remove studs and lift flanged mounting plate.

6 Pull tank unit carefully upward and outward.

7 Remove tank unit and cover hole.

Caution Tank units must be handled carefully to avoid bumping which can produce internal damage to unit.

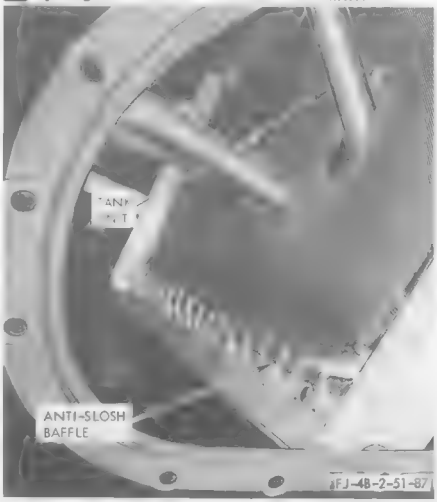
INSTALLING

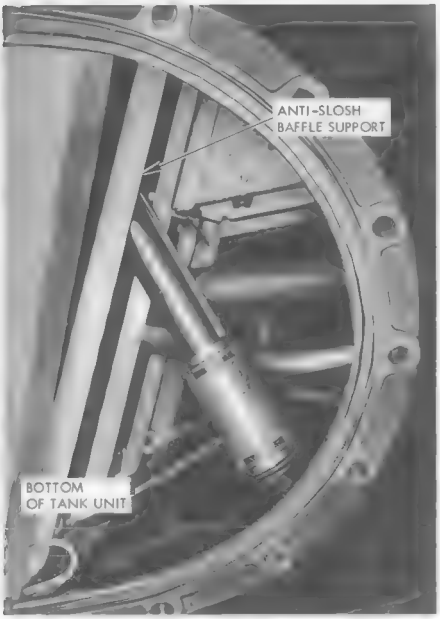
1 Coat "O" ring on bottom of tank unit with petrolatum (item 100, materials list) and lower tank unit carefully into mounting hole with lower end tilted slightly inboard and forward.

FJ-48-2-51-86

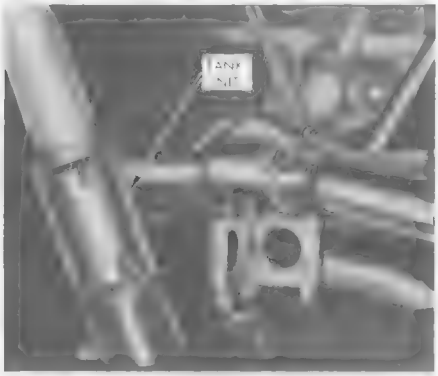
Caution Handle tank unit carefully so as not to mar or scratch the protective coating.

2 Rotate tank unit into position to pass through the opening at the aft end of the anti-slosh baffle.





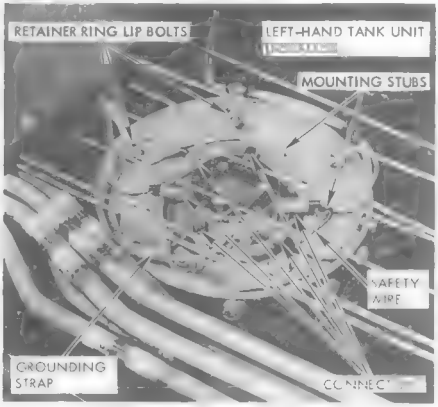
3 Seat tank unit in rubber mount at bottom of fuel cell.



Note It is imperative that tank units be properly seated in mount or fuel sloshing and excessive "G" loads will bend the unit and change its capacitance value.

FJ-4B-2-51-90

4 Place tank unit mounting plate in position over tank unit flange and line up holes with those in fuel cell. Start a mounting stud and jockey tank unit and mounting plate into position; add more studs until opposing bolts can be mounted on lip of retainer ring. When bolts and grounding strap are loosely installed, tighten bolts on retainer ring lip; then tighten tank unit studs.



5 Install safety wire (AN995F32) on tank unit mounting studs.

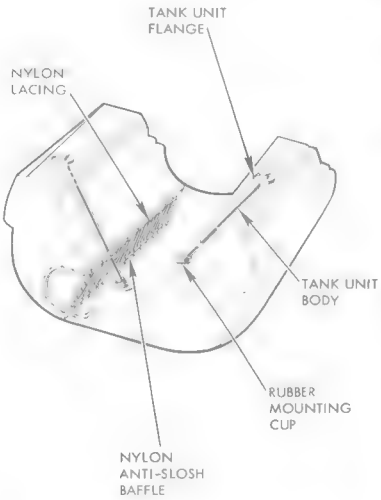
6 Install coaxial connectors and tighten grounding strap.

7 Reconnect aileron and speed brake control cables.

8 Perform an operational check of fuel quantity system. (Refer to paragraph 6-330.)

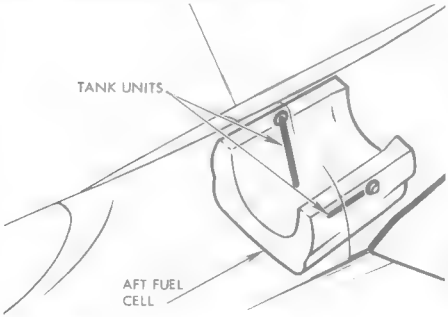
FJ-4B-2-51-88

Note No sealant of any kind is to be used in installing tank units.



Note The right-hand tank unit is mounted at a greater forward angle than the left-hand tank unit.

AFT FUSELAGE TANK UNITS



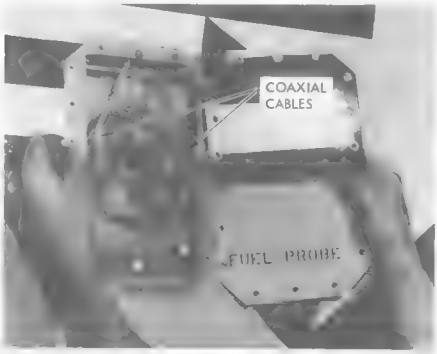
REMOVING

Note The aft fuel cell tank units are of rigid construction and do not require support mounting within the cell.

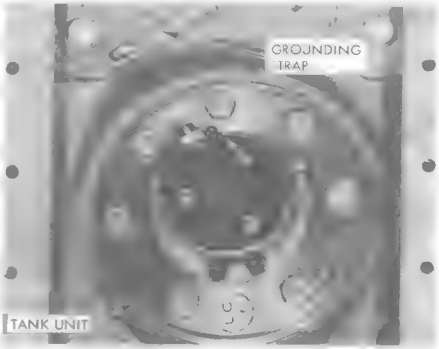
1 Remove fuel probe access cover (left- or right-hand side of fuselage) just aft of field break.

FJ-48-2-51-89

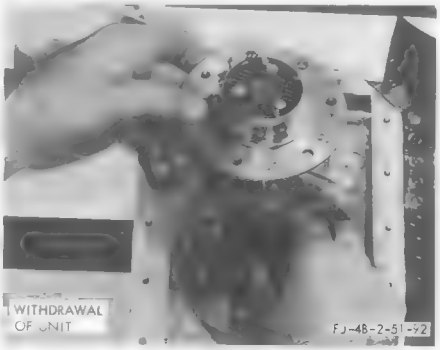
2 Remove coaxial cables and ground strap carefully from unit receptacles.



3 Remove safety wire and grounding strap from unit.



4 Remove mounting studs and lift tank unit carefully out of cell.



Caution Repair or replace coaxial leads which show any damage whatsoever. (Refer to paragraph 6-338.)

5 Cover fuel cell hole if tank unit is not to be replaced immediately.

INSTALLING

1 Remove fuel cell cover if present.

2 Coat tank unit "O" ring with a layer of petrolatum (item 100, materials list) and lower carefully into position in fuel cell, rotating as necessary to make tank unit flange fit fuselage contour.

3 Install, tighten and safety-wire tank unit mounting studs. Use AN995F32 safety wire.

4 Install coaxial cables and grounding strap on tank unit and replace cover.

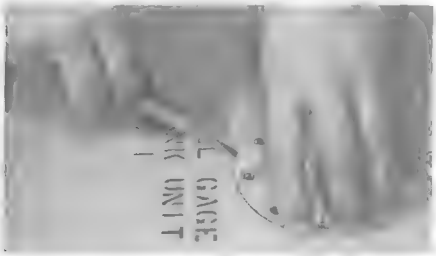
5 Perform operational check of fuel quantity system. (Refer to paragraph 6-330.)

WING TANK UNITS

REMOVING

1 Remove screws which hold tank unit in wing.

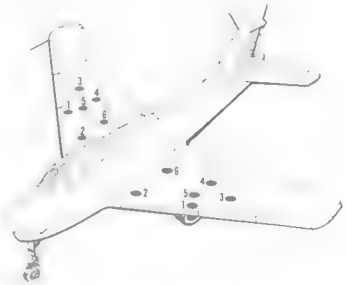
2 Insert small screwdriver into slot along edge of tank unit and pry gently to lift unit.



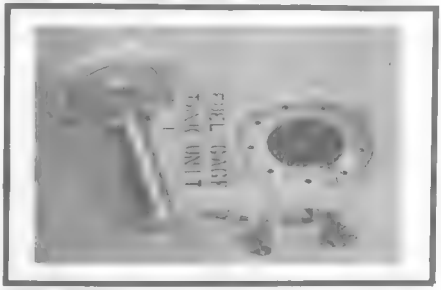
3 Lift unit out of wing and disconnect coaxial connectors.

Caution Do not allow foreign matter or dirt to fall into tank unit holes.

FJ-48-2-51-93



4 Stow wires and cover hole until replacing tank unit.



INSTALLING

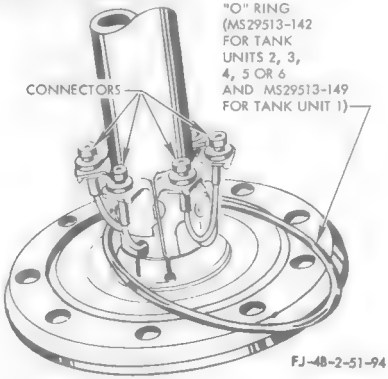
Caution Repair or replace coaxial leads which show any damage whatsoever.

1 Coat "O" ring on flange of tank unit with petrolatum (item 100, materials list) and position unit above hole.

2 Install coaxial connectors on tank unit and lower unit into hole with mounting holes aligned.

3 Install tank unit mounting screws.

4 Perform operational check of fuel quantity system. (Refer to paragraph 6-330.)



FJ-48-2-51-94

6-336. PRECAUTIONS WHEN HANDLING FUEL QUANTITY INDICATING SYSTEM WIRING.

6-337. Certain precautions must be taken when handling fuel quantity system components and wiring to assure subsequent proper operation of the system. The following is a list of instructions to be followed:

a. Always carefully recalibrate the system after replacing a tank unit or indicator-power unit. (Refer to paragraph 6-330.)

b. Refer to paragraph 6-338 for instructions on fabricating coaxial fittings when it is necessary to repair a lead rather than replace it.

c. Never force fittings to mate when more than normal pressure is required to flatten the "O" ring seal. Bent or broken fittings will result. A visual examination should reveal the discrepancy.

d. When a tool is needed to twist a fitting loose, always investigate the cause and correct before reinstalling.

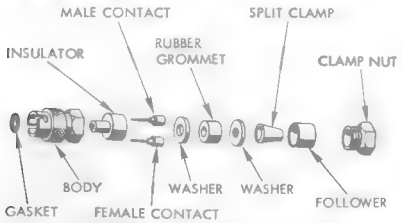
e. When using a tool to free a fitting, always grip the fitting at the end nearest the seal. Connector crimp may break if abnormal pressure is applied at the other end. Also, do not attempt to twist a connector more than 30 degrees in either direction or the approximate width of the locking slot.

f. When installing the center electrical connector in the indicator-power unit, make certain that the 12-pin miniature cable connector is keyed properly with the receptacle on the power unit. Push on the connector carefully. Improper mating of this connector will cause pins to break or push out of the cable connector.

6-338. FABRICATING FUEL QUANTITY INDICATING SYSTEM CABLE CONNECTORS AND BULKHEAD CONNECTORS.

CABLE CONNECTORS

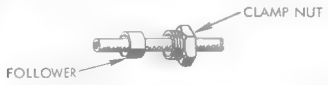
It is strongly recommended that the following steps be carried out explicitly in their respective order as shown, so that maximum efficiency will be attained when the fitting and cable is used to perform its prescribed function.



Note For cutting jacket and dielectric of cable, use a razor blade or suitable stripper. Do not use cutting pliers.

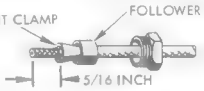
FJ-48-2-51-96

1 Slide clamp nut and follower over wire.



2 Cut off outer insulator 5/16 inch from end of wire.

3 Place split clamp over outer insulator, so that the forward end of clamp is in line with the end of the outer insulation of the wire or 5/16 inch from end of wire. Slide follower over split clamp.



4 Fan braid back over the split clamp. Trim excess braid.

5 Place first washer, rubber grommet and second washer over inner dielectric and up to split clamp.

6 Cut off inner dielectric 1/16 inch from end of wire. Tin the center conductor of the wire.

7 Solder tinned wire into contact pin, then slide insulator over pin.

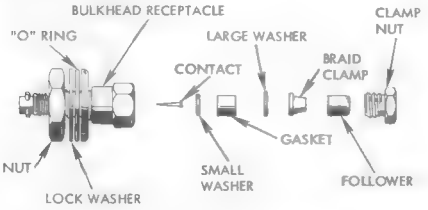
8 Insert cable assembly into connector, push firmly in place and tighten nut until it is flush with connector. After assembly, place front gasket in position shown

Note Completed assembly shown in cross sectional view with all parts in their respective positions.

FJ-48-2-51-97

BULKHEAD CONNECTORS

Fabricating instructions for these bulkhead receptacles are identical to those used on cable connectors except that the contact after fabrication to cable is inserted into a Teflon insulated sleeve not shown which is positioned inside the receptacle.



Note Since the Teflon insulated sleeve fits the receptacle loosely, it must be handled carefully to prevent its dropping out during fabrication.

FJ-48-2-51-189

6-339. CHECKING FUEL QUANTITY INDICATING SYSTEM COAXIAL CABLES. Hi-pot and megger tests of system wiring are required whenever cables have been disturbed or refabricated. Both ends of the cables must be disconnected during these tests. The tests are performed as follows:

a. Hi-pot the cable assembly with 1000 volts rms, 60 cycles, connecting one test lead to the conductor and the other to the connector body. No failure should occur within 10 seconds.

WARNING

Do not perform hi-pot tests on airplane cables when airplane contains fuel. Also, make certain that both ends of the cables are free of equipment and that ground handling personnel are clear of airplane during these tests. Extensive damage can be done by explosion of vapor from electrical arcing and personnel can be subjected to severe shock.

b. Perform a megger check between the inner conductor and the shell of the connector. This resistance should not be less than 500 megohms.

6-340. FUEL QUANTITY SYSTEM INDICATOR-POWER UNIT.

6-341. The indicator-power unit is located on the instrument panel and consists of the following: a hermetically-sealed indicator, a capacitance-sensing bridge circuit, an electronic bridge amplifier, a small two-phase motor, a gear train (between the motor and indicator pointer), a test switch provision to deflect the indicator pointer and "EMPTY" and "FULL" adjustment screws. The indicator pointer is driven over a 320-degree scale which is graduated from 0 to 6000 pounds. The gear train which connects the motor to the pointer holds the pointer at the position at which the capacitance-sensing bridge circuit seeks a null, or the position where the bridge is balanced, indicating actual compensated fuel quantity. The bridge amplifier is incorporated to increase the sensitivity of the bridge and to provide, simultaneously, current to drive the motor to the bridge balancing position of the balancing rheostat in the amplifier. The indicator-power unit receives its signal (capacitance) from the wing and fuselage cell tank units and is powered by 115 volts ac from the primary "C" phase bus through the FUEL QUANTITY fuse located on the right-hand rear vertical console. The unit contains "EMPTY" and "FULL" adjustments which are accessible on the back side when three locking screws on the back of the unit are loosened and the back ring is rotated. (See figure 6-67.)

6-342. FUEL GAGE READS SELECTOR SWITCH.

6-343. The fuel gage selector switch (FUEL GAGE READS), located on the pilot's instrument panel adjacent to the indicator, provides a two-way check of the indicating system. In the "TOTAL" or up position, the FUEL GAGE READS switch permits measurement of total fuel. In the "SUMP" or center position, the indicator reads only forward fuselage fuel, while the momentary down or "CHECK" position checks the indicator by deflecting the pointer counterclockwise. The "CHECK" position, in effect, unbalances the bridge circuit in the amplifier, thus causing the pointer to deflect. However, should the amplifier or gear drive fail, the pointer would not react in this manner. The circuitry, which allows sump or forward fuselage fuel to be read, includes the selector relay, connected so as to substitute simulated empty capacities in place of the wing tank units, and the aft fuselage tank units. (See figure 6-65.)

6-344. LOW FUEL WARNING LIGHT.

6-345. A red, low fuel warning light is installed on the pilot's instrument panel to give an obvious indication of low fuel. The light is controlled by a float switch mounted within the forward fuselage fuel cell. It receives its power from the 28-volt d-c primary bus and is protected by a 5-ampere circuit breaker (INST. LIGHTS) located on the right-hand forward console. The low fuel warning float switch, when actuated, energizes the low fuel sensing relay which, in turn, controls the low fuel warning light. Dimming of the low fuel warning light is provided by the warning light dimming relay which is actuated by turning on the instrument lights rheostat.

6-346. LOW FUEL WARNING FLOAT SWITCH.

6-347. The low fuel warning float switch is preadjusted before installation to actuate at 950 (±50) pounds of fuel. When the forward fuselage fuel cell contains more than 950 pounds of fuel, the float switch is in the "wet" position and there should be no continuity through the switch. When the fuel cell is empty or contains less than 950 pounds of fuel, the float switch is in the "dry" position and there should be continuity through the switch. (See figure 6-18.)

Note

The low fuel warning light will flash on intermittently or remain on during violent maneuvers due to fuel sloshing within the cell or due to excessive "G" loads applied to the float switch arm, causing it to actuate.

Figure No. 6-70 deleted.

6-347A. TROUBLE SHOOTING LOW FUEL WARNING SYSTEM.

TEST EQUIPMENT: D-C voltmeter.
Ohmmeter.

SYSTEM CONDITIONS: INST LIGHTS circuit breaker engaged.
External power on airplane.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
LOW FUEL WARNING LIGHT ILLUMINATED (FULL FUEL IN AIRPLANE).			
Defective float switch or associated wiring.	Check between test point EQG and ground.	Zero ohms.	Continue trouble shooting.
		Other than zero ohms.	Replace defective float switch or associated tank internal wiring.
Defective LOW FUEL SENSING relay or associated wiring.	Check between test point EQD and ground and between test points EQD and EQE.	28 volts dc.	Replace defective LOW FUEL SENSING relay.
		28 volts dc at test point EQD only.	Replace defective LOW FUEL SENSING relay if zero ohms or infinite ohms exist between test points EQD and EQE.
		Zero volts at test point EQD.	Perform wire segment continuity check to test point PBC and replace defective circuit breaker or wire segment as required.
Defective warning light ground circuit.	Disengage pressure disconnect No. 12.	None.	If warning light extinguishes, replace defective LOW FUEL SENSING relay. If light does not extinguish, repair or replace shorted wire segment between test points EQH and EQJ.
WARNING LIGHT WILL NOT ILLUMINATE. (LESS THAN 950 POUNDS OF FUEL IN CELL.)			
Faulty warning light lamps.	Actuate warning light test switch.	None.	If light illuminates, continue trouble shooting. If light does not illuminate, replace defective lamps.
Defective power input.	Check between test point EQL and ground.	28 volts dc.	Continue trouble shooting.
		Zero volts.	Replace defective power wire segment or circuit breaker.

PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
WARNING LIGHT WILL NOT ILLUMINATE. (LESS THAN 950 POUNDS OF FUEL IN CELL. (Cont)			
Defective warning light ground circuit or LOW FUEL SENSING relay.	Check between test point EQH and ground. **Note** If warning light dimming circuit is energized, use test point EQK instead of EQJ. Utilize listed remedies and if circuit fault is not found, perform wire segment continuity check through warning light dimming relay to test point EQJ and replace defective warning light dimming relay or wire segment as required.	Zero volts.	Repair or replace open wire segment between test points EQH and EQJ.
		28 volts dc.	Continue trouble shooting.
	Check between test point EQF and ground.	Zero volts.	Replace defective wire segment to test point EQH.
		28 volts dc.	Replace defective LOW FUEL SENSING relay.
Defective low fuel warning float switch.	Check between test points EQD and EQE.	28 volts dc.	Replace defective low fuel warning float switch.
		Zero volts.	Replace defective LOW FUEL SENSING relay.

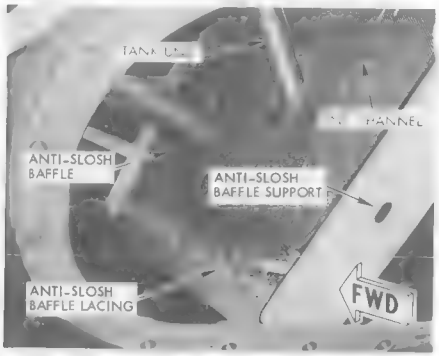
6-348. REMOVING AND INSTALLING LOW FUEL WARNING FLOAT SWITCH.

REMOVING

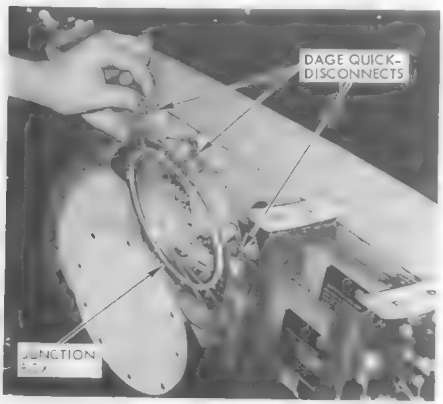
1. Defuel airplane. (Refer to paragraph 1-35.)
2. Gain access to inside of forward fuselage fuel cell by removing forward fuselage fuel cell access plate. (See figure 4-230.)

Caution Any fuel cell which has contained fuel is dangerous. Observe all safety precautions.

3. Unlace a portion of the anti-slosh baffle to allow a more convenient entry into the cell.

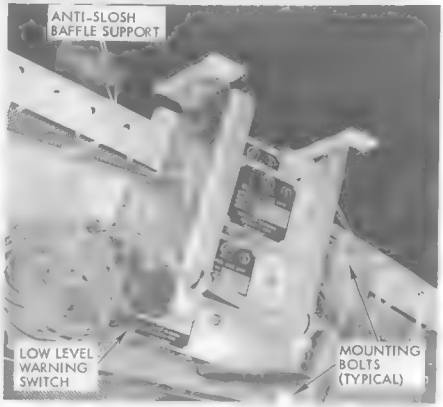


4. Remove baffle support and end channel.
5. Enter cell headfirst.
6. Disconnect Dage quick-disconnects from junction box.



FJ-4B-2-51-199

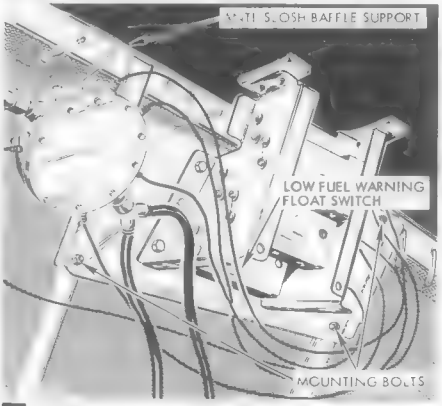
7. Remove four mounting bolts which secure float switch assembly mounting plate to anti-slosh baffle support.



8. Remove float switch assembly from cell.
9. Replace low fuel warning float switch with a like serviceable unit.

INSTALLING

1. Enter cell head first.
2. Place float switch assembly in position in line with four mounting holes on anti-slosh baffle support.
3. Install and tighten four mounting bolts.



4. Replace baffle support, end channel and anti-slosh baffle.
5. Connect five Dage quick-disconnects to junction box.
6. Install forward fuselage fuel cell access plate. (Refer to paragraph 4-230.)

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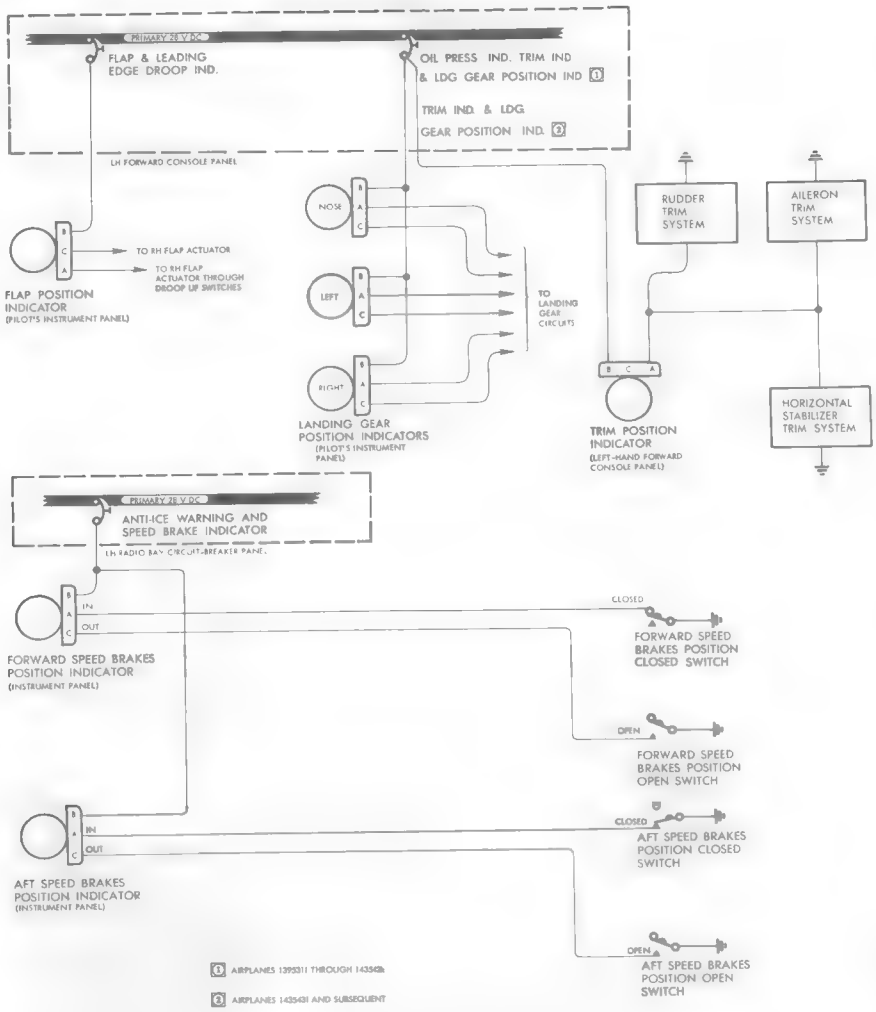

Figure No. 6-71. Position Indicating Systems

POSITION INDICATING SYSTEMS

6-349. POSITION INDICATING SYSTEMS.

6-350. The position indicating systems (figure 6-71) include the landing gear position warning and indicating system, the flap position indicating system, the flight control take-off trim position indicating system and the forward and aft speed brakes position indicating system. Each of the position indicating systems contains lock switches or microswitches located near the operation which is being performed. These switches will complete a circuit to energize a corresponding position indicator, located in the cockpit, when a certain conditional operation takes place or when a certain position of the controls has been assumed. No attempt is made to indicate the amount of linear movement or operation. The position indicating systems are so closely related with the fundamental actuation of each specific system that a complete description of each system is given in the respective system paragraph. (Refer to the Index for the page number of the system concerned.) The following paragraphs are limited to a description of the actuation and operation of the position indicators. For removing and installing the landing gear position indicators, the flap position indicator and the speed brakes position indicators, refer to paragraphs 6-7 and 6-8.

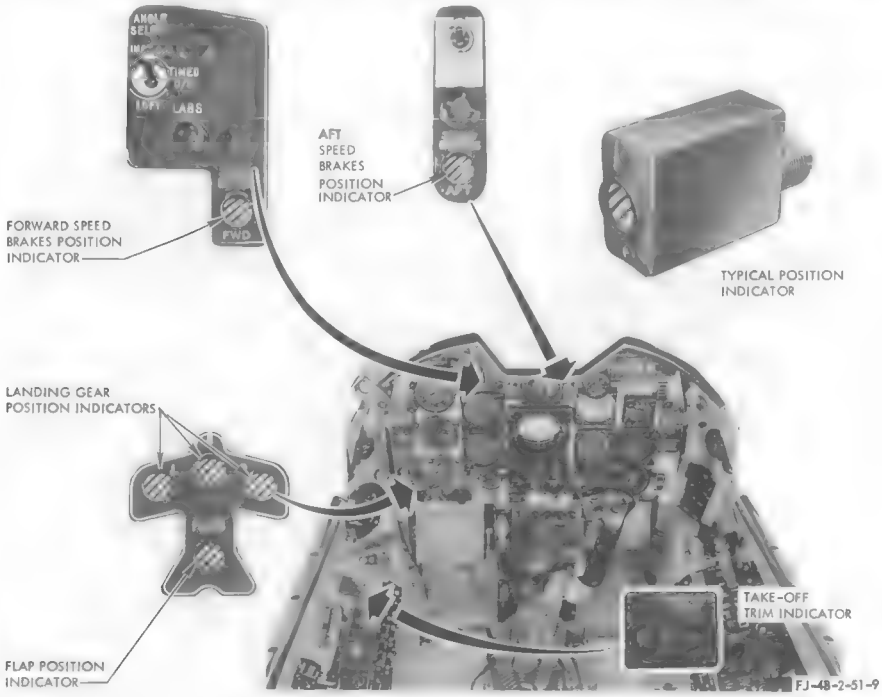

Figure No. 6-72. Position Indicators

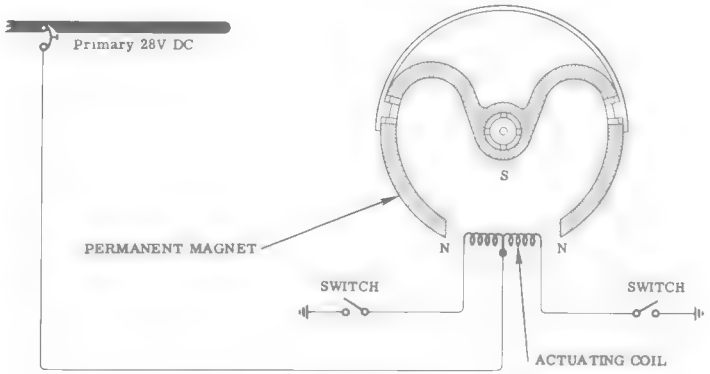

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Figure No. 6-73. Position Indicator Schematic Diagram

6-351. POSITION INDICATORS.

6-352. The position indicators (figure 6-72), with the exception of the take-off trim position indicator, are located on the pilot's instrument panel. The take-off trim position indicator is located on the forward left-hand console panel. Each indicator is a solenoid-operated, three-position (two positive and one neutral) indicator which consists of a heart-shaped, permanent magnet, a center tapped actuating coil and a scale plate attached to the magnet to provide indications. (See figure 6-73.) The indicators are hermetically sealed and are filled with helium to prevent corrosion. The neutral position of the indicator is held by spring tension and the neutral indication will be visible when the coil is de-energized or when there is no operating power. A positive indication is visible when either side of the coil is energized, overpowering the spring tension. Action at the indicators is initiated by the action which takes place at the switch. One side of each switch is at ground potential and the other side is connected to one end of the coil. The center tap of the coil receives 28-volt d-c power from the primary bus; each position indicating circuit is protected by a 5-ampere circuit breaker. Depending upon the condition that exists at the switch, one side of the coil may be energized independently of the other side to produce a magnetic field. By magnetic repulsion and attraction between the energized coil and the permanent magnet, the magnet and attaching scale are rotated. This brings the proper indication into view on the face of the indicator.

6-353. TROUBLE SHOOTING POSITION INDICATORS.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
LANDING GEAR POSITION INDICATORS		
BARBER POLE INDICATION AT ALL TIMES.		
Faulty power supply, open circuit breaker or break in wiring.	Check circuit from the circuit breaker to pin "B" on the indicator.	Eliminate open circuit or defective wiring.
Defective indicator.		Replace indicator.
Defective uplock or downlock switches in landing gear system.		Replace and adjust switches. (Refer to paragraph 3-117.)

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	LANDING GEAR POSITION INDICATORS (Cont)	
BARBER POLE INDICATION AT ALL TIMES. (Cont)		
Trouble in landing gear system.		Refer to paragraph 3-102.
"UP" INDICATION WHEN THE WHEELS ARE DOWN OR "WHEEL" INDICATION WHEN WHEELS ARE UP.		
Connections "C" and "A" are reversed at the indicator.	Check wiring to pins "C" and "A" at the indicator.	Reverse connections.
Defective indicator.		Replace indicator.
Defective indicating switches in the landing gear uplock or downlock switches.	Refer to paragraph 3-117.	Replace and adjust switches. (Refer to paragraph 3-117.)
	WING FLAP POSITION INDICATOR	
BARBER POLE INDICATION AT ALL TIMES.		
Faulty power supply, open circuit breaker or break in wiring.	Check circuit from the circuit breaker to pin "B" on the indicator.	Eliminate open circuit or defective wiring.
Defective indicator.		Replace indicator.
Defective switches in right-hand wing flap actuator or defective droop up switches (up cycle only).		Replace actuator. (Refer to paragraph 2-340.)
Failure of outboard wing droop leading edge parts.	Visually inspect right and left outboard wing droop leading edge attaching and actuating parts.	
Trouble in wing flap system.		Refer to paragraph 2-333.
"UP" INDICATION WHEN FLAPS ARE DOWN OR "DOWN" INDICATION WHEN FLAPS ARE UP.		
Connections "C" and "A" reversed at the indicator.	Check wiring to pins "C" and "A" at the indicator.	Reverse connections.
Defective indicator.		Replace indicator.
Defective switches in right-hand wing flap actuator.		Replace actuator. (Refer to paragraph 2-340.)
	TRIM POSITION INDICATOR	
BLUE INDICATION AT ALL TIMES.		
Faulty power supply, open circuit breaker or break in wiring.	Check circuit from the circuit breaker to pin "B" on the indicator.	Eliminate open circuit or defective wiring.
Defective indicator.		Replace indicator.
Defective trim switches or trouble in rudder trim, aileron trim or horizontal stabilizer trim systems.		Refer to paragraphs 2-252, 2-268 and 2-295.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
SPEED BRAKES POSITION INDICATORS		
BARBER POLE INDICATION AT ALL TIMES.		
Faulty power supply, open circuit breaker or break in wiring.	Visually check circuit from the circuit breaker to pin "B" on indicator.	Eliminate open circuit or defective wiring.
Defective indicator.		Replace indicator.
Defective speed brake microswitches.		Replace and adjust microswitches. (Refer to paragraphs 3-291 and 3-292.)
Trouble in speed brake system.		Refer to paragraph 3-258.
"IN" INDICATION WHEN SPEED BRAKES ARE OUT OR "OUT" INDICATION WHEN SPEED BRAKES ARE IN.		
Connections "C" and "A" are reversed at the indicator.	Check wiring to pins "C" and "A" at indicator.	Reverse connections.
Defective speed brake microswitches.		Replace and adjust microswitches. (Refer to paragraphs 3-291 and 3-292.)

6-354. TESTING POSITION INDICATORS.

6-355. The position indicators may be checked to see if the internal wiring is correct and if the correct indication comes into view for the operation that is being performed. All of the position indicators are operated in the same way but are modified to indicate different functions. Before testing the indicator, check the position indicator schematic (figure 6-73) and the indications (figures 6-74 through 6-77) to ascertain what indication should appear. To test the indicator, proceed as follows:

a. Connect positive lead of a 28-volt d-c power source to pin "B" of indicator.

b. Connect negative lead of power source to pin "A" of indicator.

c. Visually check that proper indication comes into view.

d. Remove negative lead from pin "A."

e. Visually check that neutral indication (power-off) comes into view.

f. Connect negative lead of power source to pin "C" of indicator.

g. Visually check that proper indication comes into view.

h. Remove both negative and positive leads from indicator.

6-356. LANDING GEAR POSITION INDICATORS.

6-357. Three landing gear position indicators, located at the lower left-hand corner of the instrument panel, provide visual indications of the landing gear positions. The left and right main landing gear indicators are identified by "L" and "R," respectively, and the nose landing gear is identified by "N." When the landing gear and the landing gear doors are locked in the up position, the normally open landing gear uplock switches and door close, and lock switches in the landing gear circuit close, energizing the landing gear indicator's solenoid. The "UP" position then appears through the cutout on the face of the indicator and remains thus until circuit conditions change. Similarly, when the landing gear is locked in the down position, the downlock switches close, resulting in a miniature landing wheel coming into view. The barber-pole indication appears when the uplock and downlock switches are open or when there is no power to the indicator.

6-358. FLAP POSITION INDICATOR.

6-359. The flap position indicator, located on the instrument panel below the three landing gear position indicators, is identified with the word "FLAPS" on the refractor. Indicator action is initiated by switches located in the right-hand wing flap actuator and each of the leading edge panels. The indication in view tells whether the flaps are retracted or extended. When the flaps-up switch is closed and the droop-up switches are closed, the positive indication is "UP." A down indication is obtained by the actuation of the flaps-down switch only as the droop switches are by-passed on the down cycle. The pilot must visually check the mid-flap position in the mirror, since the indicator will show a barber pole in the mid-flap position. The barber-pole indication also appears whenever the flaps-up, droop-up or flaps-down switches are open or whenever there is no power to the indicator.

6-360. TAKE-OFF TRIM POSITION INDICATOR.

6-361. The take-off trim position indicator, located on the forward left-hand console, is wired into the trim circuits of the horizontal stabilizer, the ailerons and the rudder. An "IN" indication is provided only for the control surface that is being trimmed and when that control surface is passing through the take-off trim position. When no power is available to the indicator, a solid blue with no lettering is visible.

6-362. FORWARD AND AFT SPEED BRAKES POSITION INDICATORS.

6-363. Two identical position indicators are installed on the top center of the instrument panel. One indicator shows the position of the forward speed brakes; the other indicator, marked "AFT," shows the position of the aft speed brakes. Indications are the same for both indicators. The two positive indications are "IN" and "OUT" and the neutral position is a barber-pole indication.

6-364. REMOVING TAKE-OFF TRIM POSITION INDICATOR. To remove take-off trim position indicator, proceed as follows:

a. Make sure that all power to the instruments is off.

b. Remove lighting fixture from take-off trim position refractor.

c. Remove refractor.

d. Unfasten left-hand console panel by turning eight console fasteners counterclockwise.

e. Lift console panel up far enough to gain access to the take-off trim position indicator.

f. Remove electrical connector from indicator.

g. Remove two indicator mounting screws and withdraw the indicator from under the panel assembly on which is it mounted.

6-365. INSTALLING TAKE-OFF TRIM POSITION INDICATOR. To install the take-off trim position indicator, proceed as follows:

a. Position the indicator on its mounting panel and secure with two indicator mounting screws.

b. Make electrical connection.

c. Return the left-hand console panel to its mounted position and secure by turning eight console fasteners clockwise.

d. Install refractor.

e. Secure refractor with lighting fixture.

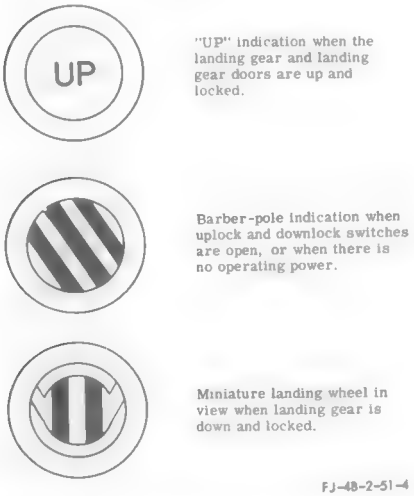


Figure No. 6-74. Landing Gear Position Indications

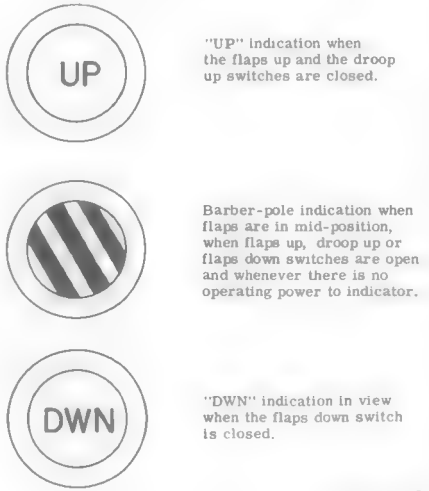
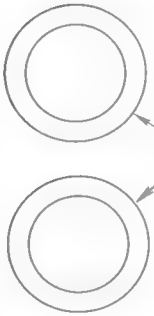


Figure No. 6-75. Flap Position Indications

"IN" indication in view momentarily after each control surface is trimmed. (Trim switch in position for that control surface.)



Plain blue indication in view when the control surface is not in trim or when there is no operating power to the indicator.

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Figure No. 6-76. Take-off Trim Position Indications

"IN" indication in view when speed brakes position closed switches are energized.

Barber-pole indication when position open and position closed switches are not energized or when there is no operating power to the indicator.

"OUT" indication in view when speed brakes position open switches are energized.

FJ-48-2-51-7

Figure No. 6-77. Forward and Aft Speed Brakes Position Indications

HYDRAULIC PRESSURE INDICATING SYSTEMS

6-366. HYDRAULIC PRESSURE INDICATING SYSTEMS.

6-367. The hydraulic pressure indicating systems provide a continuous pressure indication of the No. 1 and No. 2 flight control systems and the utility hydraulic system. The indicating systems consist of three remote pressure transmitters, one located in each of the system pressure lines, and a hydraulic pressure selector switch and a dual pointer indicator, both located on the pilot's instrument panel. The systems are powered by 26-volt, 400-cycle, single-phase alternating current from the 26-volt, single-phase a-c bus. The circuit is protected by a one-ampere fuse [HYD PRESS IND on airplanes 139531i through 143542k (HYD & OIL PRESS IND on airplanes 143543l and subsequent)] located on the right-hand rear vertical console. (See figure 6-78.)

6-368. FUNCTION OF HYDRAULIC PRESSURE INDICATING SYSTEMS.

6-369. A pressure transmitter of the Bourdon tube-type is located in each hydraulic system pressure line. Expansion and contraction of the Bourdon tube is transmitted, by mechanical linkage, to the rotor of the transmitter synchro. The pressure transmitter synchro transmits an electrical signal, through interconnecting wiring, to the receiving synchro within the indicator. The receiving synchro's rotor is mechanically linked to the indicator pointer. The hydraulic pressure indicator contains two synchros mechanically attached to two separate pointers. When the HYD PRESS SELECTOR switch is in the "NO. 1 & NO. 2 FLT. CONT" position, the pointers indicate the pressure in each system independent of each other. When the HYD PRESS SELECTOR switch is in the "UTILITY" position, the synchros are connected electrically in parallel; therefore, the pointers act as one. (See figure 6-79.)

Note

If at any time during indicator operation the a-c power is interrupted, the pointers will remain fixed.

6-370. TROUBLE SHOOTING HYDRAULIC PRESSURE INDICATING SYSTEMS.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
INDICATOR POINTER SLUGGISH BUT TRIES TO FOLLOW THE TRANSMITTER.		
No power on one rotor.	Check out connections to pin "B" on the transmitters and pins "B" and "E" on the indicator.	Connect wiring as necessary.
Defective indicator.		Replace indicator. (Refer to paragraphs 6-7 and 6-8.)
POINTER 180 DEGREES IN ERROR.		
No power on one rotor.	Check out connections to pin "B" on the transmitters and pins "B" and "E" on the indicator.	Connect wiring as necessary.
POINTER SWINGS IN LIMITED ARC ACROSS TOP OF DIAL.		
Open transmitter ground lead.	Check for loose connection to pin "A."	Connect as necessary.
Reversed power leads.	Check out connections to pin "B" on the transmitters and pins "B" and "E" on the indicator.	Connect as necessary.
POINTER SWINGS IN LIMITED ARC ACROSS BOTTOM OF DIAL.		
Open indicator ground lead.	Check for loose connection to pin "A" at the indicator.	Connect as necessary.

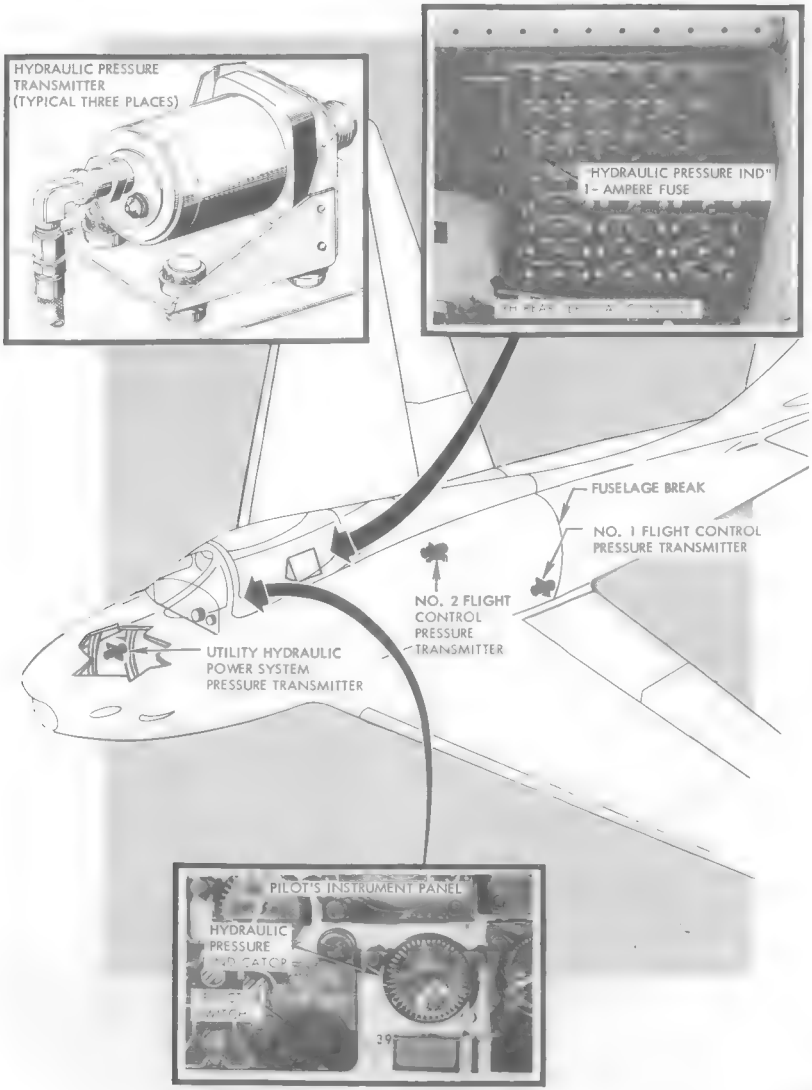

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Figure No. 6-78. Hydraulic Pressure Indicating System Unit Location

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
MOST OF POINTER SWINGS AT SIDE OF DIAL; SQUEAL HEARD INSIDE INDICATOR.		
Short circuit between power and stator lead; reversed connection between power and stator lead.	Check wiring in circuit and power connections to pin "B" on the transmitters and pins "B" and "E" on the indicator. Check stator connection from transmitters to indicator. Pin "A" is the ground lead on both the indicator and the transmitters.	Connect or reverse connections as necessary.
MOST OF POINTER SWINGS AT SIDE OF DIAL; NO SQUEAL HEARD INSIDE INDICATOR.		
Open stator lead.	Check wiring from pins "C" and "D" on the transmitters to pins "C" and "D" on the indicator. (See figure 6-79.)	Make connections as necessary.
POINTER MAKES COMPLETE ROTATION IN REVERSE DIRECTION.		
Reversed stator leads.	Check all wiring from the stators at the transmitters to the stators of the indicator. (See figure 6-79.)	Reverse leads as necessary.
SLIGHT OR NO MOVEMENT OF POINTER.		
Short circuit between stator leads.	Check stator wiring connections. (See figure 6-79.)	Eliminate short.
Short circuit between power leads.	Check power leads to pin "B" on the transmitters and to pins "B" and "E" on the indicator.	Eliminate short.
Defective indicator.		Replace indicator. (Refer to paragraphs 6-7 and 6-8.)
Defective transmitter.		Replace transmitter.
Clogged or dirty hydraulic pressure lines to the transmitter.	Remove hydraulic pressure transmitter. Examine pressure fittings and open end of line.	Remove dirt and/or obstruction.
BELOW SCALE DEFLECTION OF POINTER (PRESSURE APPLIED TO TRANSMITTER).		
Pressure connection made to vent port instead of to pressure port of transmitter.	Visually check pressure connection.	Make correct pressure connection.
POINTER SPINS.		
Power lead reversals, power and stator lead short circuit or reversal, or intermittent contact.	Visually check out all wiring. (See figure 6-79.) Check for continuity and shorts.	If wiring is correct, replace the indicator. (Refer to paragraphs 6-7 and 6-8.)
Defective indicator.		Replace indicator. (Refer to paragraphs 6-7 and 6-8.)
LOW INDICATION.		
Defective indicator.		Replace indicator.
Trouble in hydraulic systems.		Refer to trouble shooting chart in paragraph 3-7.

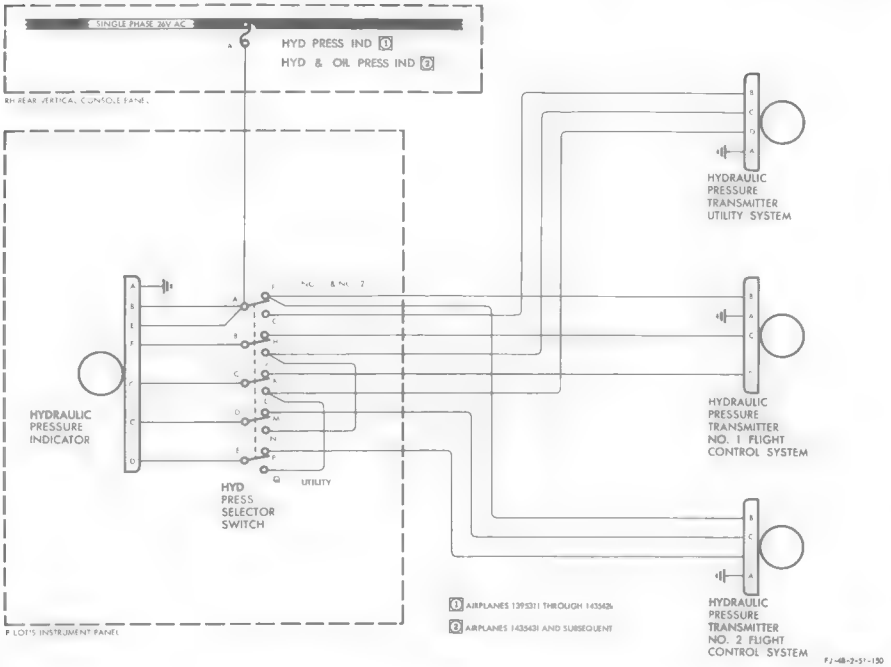


Figure No. 6-79. Hydraulic Pressure Indicating System Schematic

6-371. HYDRAULIC PRESSURE TRANSMITTERS.

6-372. The synchro-type hydraulic pressure transmitters are located in the pressure lines of the three hydraulic pressure systems. Each transmitter consists of a Bourdon tube pressure measuring device and a synchro transmitting unit. Hydraulic pressure is connected directly to the tube through a pressure fitting. When the deflections of the Bourdon tube rotate the energized rotor of the transmitter synchro, the positioning is transmitted electrically to the energized rotor of the indicator synchro.

6-373. HYDRAULIC PRESSURE SELECTOR SWITCH.

6-374. A two-position hydraulic pressure selector switch (HYD PRESS SELECTOR) is located adjacent to the hydraulic pressure indicator on the pilot's instrument panel. When the HYD PRESS SELECTOR switch is in the "NO. 1 & NO. 2 FLT. CONT" position, the dual pointers on the hydraulic pressure indicator marked "1" and "2" indicate their respective flight control system pressures. When the HYD PRESS SELECTOR switch is placed in the "UTILITY" position, both the indicator synchros are connected in parallel to the utility transmitter synchro and the two pointers align to indicate utility pressure.

6-375. HYDRAULIC PRESSURE INDICATOR.

6-376. The remote indicating synchro-type hydraulic pressure indicator is located at the bottom center of the pilot's instrument panel. The indicator contains two separate synchros, the rotors of which are geared to two separate pointers on the indicating dial. The system pressures indicated are dependent upon the position of the hydraulic pressure selector switch. (Refer to paragraph 6-373.) The indicating scale range is from 0 to 4000 psi, while the indicating pointer moves through 320 circular degrees.

MISCELLANEOUS INSTRUMENTS

6-377. MISCELLANEOUS INSTRUMENTS.

6-378. The accelerometer, the clock and the cabin pressure altitude indicator are instruments which are not essential to flight, but which provide very useful information during flight. The statistical accelerometer is installed to record the number of times, during its service life, that the airplane equals or exceeds four different preset acceleration levels. The accelerometer and the clock are located on the pilot's instrument panel, the cabin pressure altitude indicator is located on the right-hand close-out panel, adjacent to the instrument panel, and the statistical accelerometer is located in the right-hand radio bay. For removal and installation of the accelerometer and the clock, refer to paragraphs 6-7 and 6-8.

6-379. ACCELEROMETER.

6-380. The accelerometer gives a continuous indication of the acceleration of the airplane along its vertical axis. The instrument measures the load acting on the structure of the airplane and not the drag load resulting from aerodynamic resistance to flight. The acceleration is indicated in G's on a dial graduated from −5 to +10. Three pointers are visible on the face of the indicator. The main pointer gives a continuous indication of vertical acceleration. When the lift of the airplane wing is equal to the weight of the airplane, the pointer will indicate +1 G. Such a condition exists in unaccelerated level flight. Two additional pointers are provided and indicate the maximum positive acceleration and maximum negative acceleration for any one flight or maneuver. These pointers will remain fixed at their maximum indications until the PUSH TO SET knob, located at the lower left-hand corner of the indicator case, is pushed to return the pointers to their normal positions.

CAUTION

Before installing the accelerometer, make certain the locking screw, located on the rear of the case, is in the free position.

6-381. TROUBLE SHOOTING ACCELEROMETER.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
POINTER LOCKED IN POSITION.		
Vibration of instrument is below minimum requirements.	Check instrument for restriction. Energize instrument panel vibrator. Gently tap accelerometer.	Remove restriction. If sticking persists, replace accelerometer.
Excessive internal friction.		Replace accelerometer.
POINTER VIBRATES.		
Excessive vibration of internal mechanism due to insecure mounting.	Check instrument panel for worn mounting holes and screws or loose mounting adapters.	Replace screws. Re-cement mounting adapters. Replace accelerometer.
SCALE ERROR GREATER THAN ±0.25 G TOLERANCE.		
Defective indicator.		Replace accelerometer.

6-382. TESTING ACCELEROMETER. Before installation the accelerometer should be tested for static scale error. This error can be detected by holding the accelerometer in different positions and observing the indications. To test the accelerometer, proceed as follows:

a. Hold the plane of the dial vertical with the +5 G graduation at the top and release the auxiliary pointers. The instantaneous pointer should indicate +1 (±0.25) G.

b. Hold the dial horizontal and release the auxiliary pointers. The instantaneous pointer should indicate 0 (±0.25) G.

c. Hold the dial vertical with the +5 G graduation at the bottom and release the auxiliary pointers. The instantaneous pointer should indicate −1 (±0.25) G.

6-383. CLOCK.

6-384. A standard 8-day, 12-hour dial clock is installed on the pilot's instrument panel. The clock, conventional in operation, is wound by turning the knob clockwise and is set by pulling out and turning the knob as required. When setting the clock, pull out the knob and hold firmly since the knob is spring-loaded to return to the winding position. No attempt should be made

to service the clock; if clock is defective, replace with a serviceable unit.

6-385. CABIN PRESSURE ALTITUDE INDICATOR.

6-386. The cabin pressure altitude indicator, located on the right-hand close-out panel to the right of the instrument panel, is a sensitive altimeter used to measure the cabin pressure altitude. The instrument contains a sensitive diaphragm which expands or contracts with changes in cabin pressure. The rear of the indicator case is vented to the pressurized area. The cabin pressure is converted to feet of altitude and indicated on the dial which is graduated from 0 to 50 (times 1000) feet, or a range of 0 to 50,000 feet.

Note

The cabin pressure altitude indicator should be removed or the vent capped if the airplane is to undergo a cabin pressurization test on the ground. Failure to do so will result in damage to the instrument due to excessive pressure.

6-387. TROUBLE SHOOTING CABIN PRESSURE ALTITUDE INDICATOR.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
INDICATOR INOPERATIVE.		
Cabin pressure vent clogged.	Visually examine the opening on the rear of the indicator.	Remove obstruction.
INCORRECT INDICATION.		
Slow leak in indicator diaphragm.		Replace indicator.

6-388. TESTING CABIN PRESSURE ALTITUDE INDICATOR. To test the cabin pressure altitude indicator, proceed as follows:

a. Place indicator in a bell jar with another cabin pressure altitude indicator whose performance is satisfactory and whose scale errors are known or have been checked against a manometer.

b. Lower pressure in bell jar at a rate not to exceed 6000 feet per minute.

c. Take a reading at 1000, 5000 and 30,000 feet. Cabin pressure altitude indicator should indicate satisfactorily throughout its entire range. Errors should not exceed 400 feet at 1000 feet, 500 feet at 5000 feet and 1200 feet at 30,000 feet.

d. Remove indicator from bell jar.

e. Connect a source of suction to the pressure opening on the rear of the indicator. Apply suction until indicator reads 20,000 feet.

f. Pinch off suction and wait 10 seconds.

g. After 10 seconds, indication should not have changed more than 200 feet.

6-389. REMOVING AND INSTALLING CABIN PRESSURE ALTITUDE INDICATOR.

REMOVING

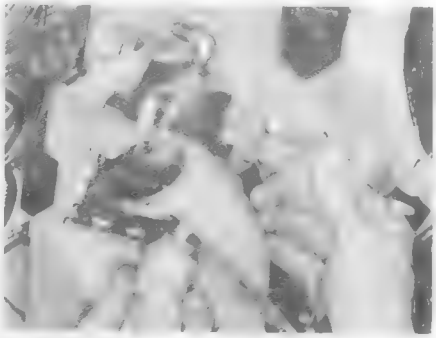
1 Remove eight screws from forward right-hand close-out panel.

2 Pull panel inboard and aft to gain access to rear of indicator.

Caution Care should be taken in pulling panel loose, as wiring to pilot's seat switch and indicator lighting fixture is connected at rear of panel.

FJ-48-2-51-161

3 Remove four indicator mounting screws and nuts and remove indicator from cutout in panel.



INSTALLING

1 Position indicator in panel and install four mounting screws and nuts.

2 Return panel to its position and install eight screws.

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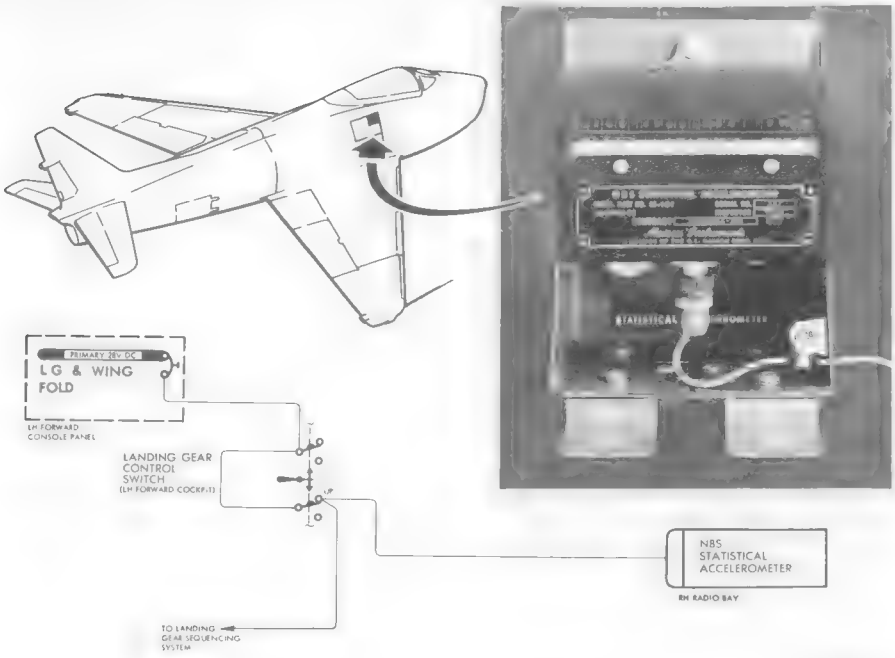
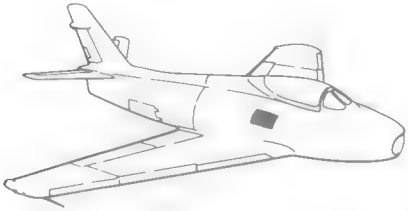


Figure No. 6-80. Statistical Accelerometer

6-390. STATISTICAL ACCELEROMETER.

6-391. The statistical accelerometer (figure 6-80) is installed in the right-hand radio bay on airplanes 139531i, 139541i, 139551i, 141449j, 141459j, 141469j, 141479j and 141489j. The statistical accelerometer obtains data on the frequency of occurrences of various "G" loads that the airplane sustains throughout its service life. Each instrument contains four counters with four associated acceleration sensing devices, arranged so that the counters record the number of times four different preset acceleration levels have been equalled or exceeded. Power for the statistical accelerometer is supplied from the primary 28-volt d-c bus and the circuit is protected by a circuit breaker (LG & WING FOLD) located on the left-hand forward console panel. Wiring is routed through the landing gear control switch to ensure operation only when the landing gear is retracted.

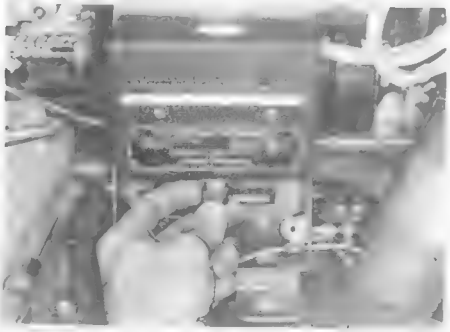
6-392. REMOVING AND INSTALLING STATISTICAL ACCELEROMETER.



REMOVING

1. Open right-hand radio bay access door.
2. Remove clamp holding the power source wire from accelerometer mounting bracket.

3. Disconnect electrical power source from accelerometer.



FJ-48-2-51-13

4. Rotate four Airloc fasteners in the accelerometer mounting bracket counterclockwise and remove the statistical accelerometer assembly from the mounting shelf.

INSTALLING

1. Position accelerometer on mounting shelf and rotate four Airloc fasteners clockwise.
2. Install clamp holding the power source wire on the accelerometer mounting bracket.
3. Connect electrical power source to the accelerometer.
4. Close and secure the right-hand radio bay door.

FJ-48-2-51-156

LIQUID OXYGEN INDICATING SYSTEM

6-393. LIQUID OXYGEN INDICATING SYSTEM.

6-394. The liquid oxygen indicating system is installed to continually monitor the level of liquid oxygen in the liquid oxygen container. The system is composed of an indicator, an amplifier, a sensing probe (located in the liquid oxygen container), a low oxygen warning light and test switch and associated electrical wiring. The indicator and the warning light test switch are located on a vertical panel at the forward end of the left-hand console. The amplifier is located on a bulkhead in the oxygen equipment compartment. See figure 6-81 for component location and figure 6-82 for electrical schematic.

6-395. FUNCTION OF LIQUID OXYGEN INDICATING SYSTEM.

6-396. The liquid oxygen indicating system is a capacitance-type system which translates the capacitance of the container probe into a quantity indication on the indicator. This capacitance translation is performed in the transistorized amplifier which is designed to balance the probe capacitance against a fixed reference. The resultant signal is amplified and used to energize a small motor which actuates the indicator pointer to the proper dial reading. The indicator also contains a precisely positioned switch, the contacts of which are closed whenever the pointer reads 0.5 ($\pm$0.1) liter or less. The closed switch contacts cause the low oxygen warning light to illuminate. A system test switch, incorporated in the spring-loaded warning light cap, returns the indicator pointer to "0" for a quick check of proper system functions.

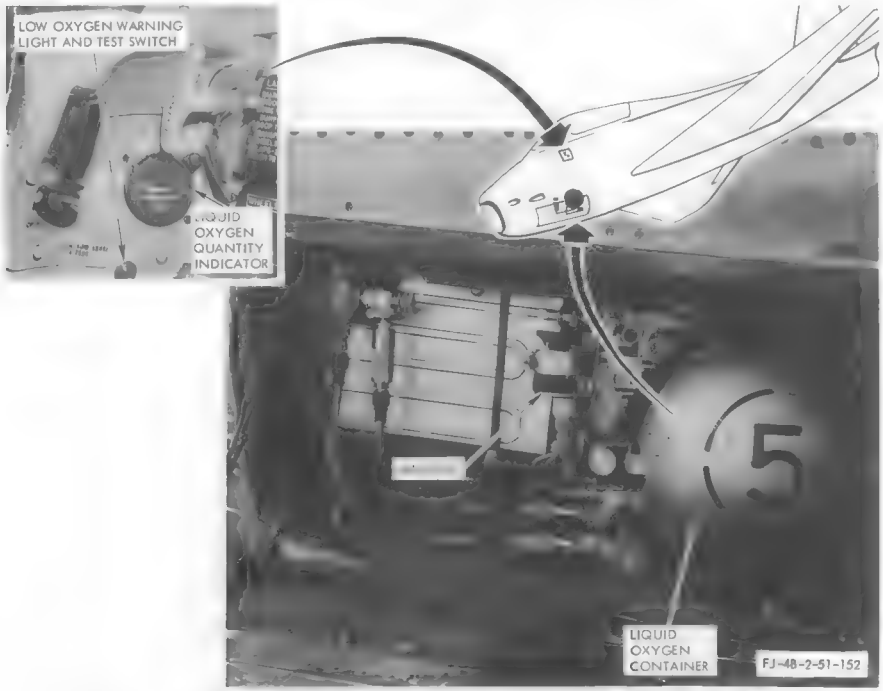
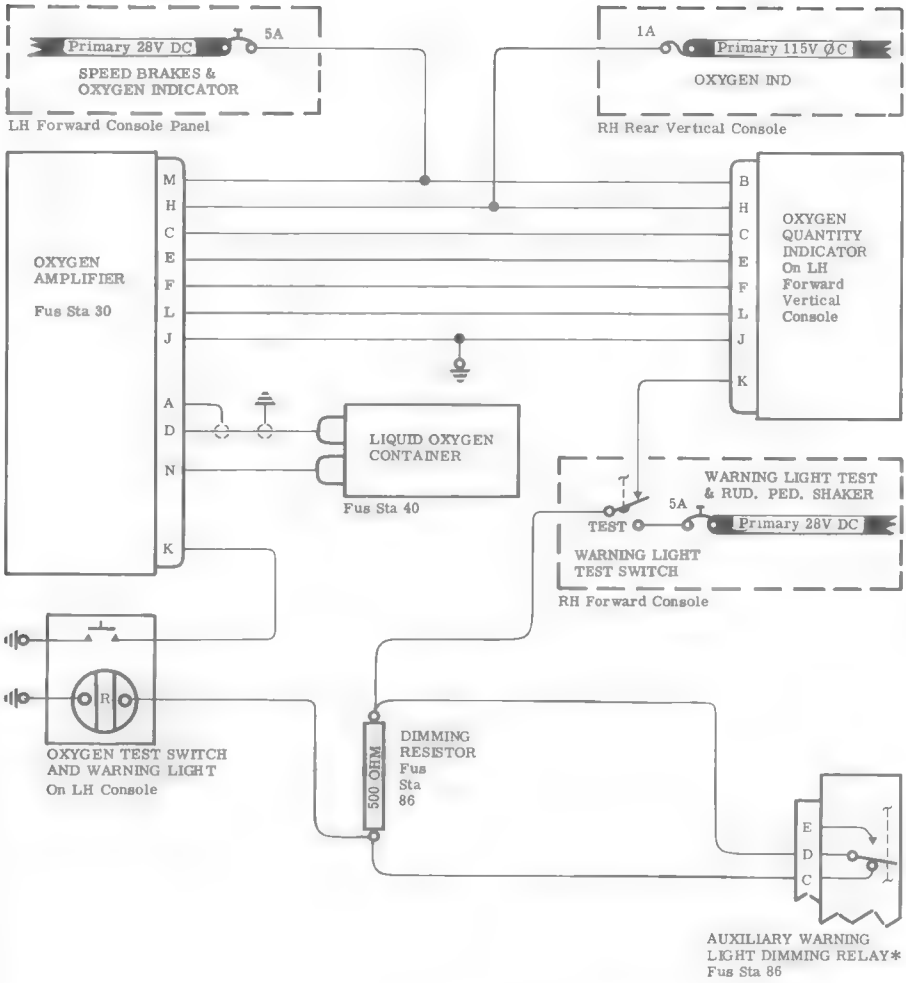


Figure No. 6-81. Liquid Oxygen Indicating System Components



*Dimming relay is energized when the instrument lights are energized.

FJ-48-2-51-162B

Figure No. 6-82. Liquid Oxygen Indicating System Schematic

6-396A. TROUBLE SHOOTING LIQUID OXYGEN INDICATING SYSTEM.

TEST EQUIPMENT: A-C voltmeter.
D-C voltmeter.
Ohmmeter.

SYSTEM CONDITIONS: External power on airplane.
SPEED BRAKES & OXYGEN INDICATOR and WARNING LIGHT TEST & RUD. PED. SHAKER circuit breakers engaged.

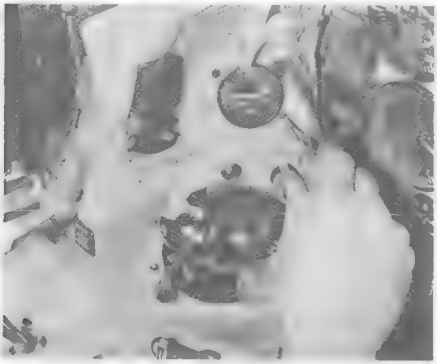
PROBABLE CAUSE	ISOLATION PROCEDURE	METER READING	REMEDY
NO INDICATION OR INCORRECT INDICATION.			
Defective power input.	Check between test points DLA, DLB and ground.	115 volts ac.	Continue trouble shooting.
		Zero volts.	Replace or repair defective power wire segment.
	Check between test points DLC, DLD and ground.	28 volts dc.	Continue trouble shooting.
		Zero volts.	Replace or repair defective power wire segment.
Defective oxygen amplifier or oxygen quantity indicator.	Refer to paragraph 6-402.	None.	Adjust or replace components as necessary.
WARNING LIGHT DOES NOT ILLUMINATE (LESS THAN 0.5 LITERS OF LIQUID OXYGEN IN CONTAINER).			
Defective lamp.	Actuate warning light test switch.	None.	If light does not illuminate, replace defective lamp. If light illuminates, continue trouble shooting.
Defective oxygen quantity indicator.	Check between test point DLE and ground.	Zero volts.	Replace defective oxygen quantity indicator.
		28 volts dc.	Continue trouble shooting.
Defective warning light circuit.	Check between test point DLF and ground.	28 volts dc.	Replace defective lamp or warning light assembly as required.
		Zero volts.	Perform wire segment continuity check between test points DLE and DLF and repair or replace defective wiring or auxiliary warning light dimming relay as required .

6-397. REMOVING AND INSTALLING LIQUID OXYGEN INDICATOR.

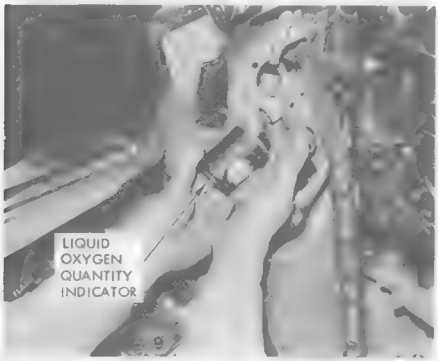
REMOVING

Warning Do not move canopy emergency release handle when removing or installing indicator. Be sure safety pin with red streamer is in handle when working in this area.

1 Loosen lower right-hand screw as shown and remove indicator from panel.



2 Disconnect electrical connector from rear of indicator.



INSTALLING

Reverse sequence of removal procedure for installation of this unit.

FJ-4B-2-51-153

6-398. REMOVING AND INSTALLING LIQUID OXYGEN INDICATING SYSTEM AMPLIFIER.

REMOVING

1 Gain access to oxygen compartment by lowering oxygen access door on forward lower left side of fuselage.

2 Loosen Marman clamp on amplifier and slide amplifier down to disengage from clamp.

3 Remove electrical quick-disconnect from amplifier by twisting counterclockwise.

INSTALLING

Reverse sequence of removal procedure for installation of this unit.

FJ-4B-2-51-154

6-399. REPLACING LIQUID OXYGEN INDICATING SYSTEM PROBE. The liquid oxygen indicating system probe is contained in the liquid oxygen container and must be replaced with the container. For instructions on replacing the container, refer to paragraph 4-126.

6-400. REPAIRING LIQUID OXYGEN INDICATING SYSTEM ELECTRICAL CABLES. The system contains several small gage coaxial cables which are delicate to handle and maintain. Extreme care must be exercised in fabricating these cables or electrical breakdown and leakage will result. For fabricating instructions, refer to paragraph 6-338. After fabricating, all cables should be megger tested (100 megohms minimum) and hi-pot tested (1000 volts rms, 60 cycles, for 10 seconds without breakdown). Any cables failing to meet these requirements must be replaced.

6-401. OPERATIONAL CHECK OF LIQUID OXYGEN INDICATING SYSTEM. To perform an operational check on the system, proceed as follows:

a. Connect an external 28-volt d-c power supply to the airplane.

Note

Place the D.C. POWER switch in "OFF" position while servicing the airplane with external power.

b. Engage SPEED BRAKES & OXYGEN INDICATOR and WARNING LIGHT TEST & RUD. PED. SHAKER circuit breakers.

c. Read liquid oxygen quantity on indicator. Reading should correspond with actual quantity in container.

d. Push and maintain pressure on cap of low oxygen warning light. Indicator pointer should move toward "0" on dial and, when pointer reaches 0.5 ($\pm$0.1) liter, low oxygen warning light should illuminate. Pointer should continue to drop until it reaches "0." Removing pressure from test switch on warning light cap should cause the indicator pointer to move back to show actual quantity.

6-402. ALTERNATE CHECK OF LIQUID OXYGEN INDICATING SYSTEM. An alternate check of the liquid oxygen indicating system can be performed with the assistance of an accurate variable capacitor such as a Simmonds Model 387001 capacitor or equivalent. This check provides an indication of the over-all accuracy of the indicating system, except the accuracy of the probe. To perform the check, proceed as follows:

a. Remove electrical leads from liquid oxygen container and connect leads to Simmonds Model 387001 variable capacitor or equivalent.

b. Connect an external 28-volt d-c power supply to airplane.

Note

Place the D.C. POWER switch in "OFF" position while servicing the airplane with external power.

c. Engage SPEED BRAKES & OXYGEN INDICATOR and WARNING LIGHT TEST & RUD. PED. SHAKER circuit breakers.

d. Set variable capacitor to 63.5 mmf and read indicator. If necessary, adjust Simmonds variable capacitor slightly until the liquid oxygen indicator reads "0." Capacitance limits for "0" reading are from 62.7 to 64.3 mmf.

e. Set variable capacitor so that liquid oxygen indicator reads "1" liter. Capacitance limits are 68.5 to 70.1 mmf.

f. Set variable capacitor so that liquid oxygen indicator reads "2" liters. Capacitance limits are 74.3 to 75.9 mmf.

g. Set variable capacitor so that liquid oxygen indicator reads "3" liters. Capacitance limits are 80.1 to 81.7 mmf.

h. Set variable capacitor so that liquid oxygen indicator reads "4" liters. Capacitance limits are 85.9 to 87.5 mmf.

i. Set variable capacitor so that liquid oxygen indicator reads "5" liters. Capacitance limits are 91.7 to 93.3 mmf.

j. Reduce variable capacitor quickly to 63.5 mmf. Indicator pointer should traverse the entire scale from "5" to "0" liters in 15 seconds or less. Low oxygen warning light should come on at 0.5 ($\pm$0.1) liter.

Note

- Pressing test switch will accomplish same reaction.
- Disregard warning light illumination on up scale pointer travel.

k. Pull SPEED BRAKES & OXYGEN INDICATOR and WARNING LIGHT TEST & RUD. PED. SHAKER circuit breakers and remove external power.

Note

After completion of alternate check, make certain that probe leads are properly reconnected to probe and that system shows proper dial indication when re-checked.

6-403. BENCH CHECKING LIQUID OXYGEN CONTAINER. In order to perform functional tests of the complete liquid oxygen indicating system, it will be necessary to remove the liquid oxygen container from the airplane since electrical test characteristics differ on this unit with respect to the amplifier and indicator. Electrical test requirements of the liquid oxygen container include a dielectric strength (hi-pot) test, an insulation resistance (megger) test and a unit capacitance test. Hi-pot and megger tests are performed by placing test cables between the center pin on each coaxial connector and ground and then between the center pins. In the hi-pot test (500 volts rms, 60 cycles), breakdown should not occur within 10 seconds. In the megger test, resistance for each point checked should not be less than 20 megohms. If the container fails to pass either the

hi-pot test or megger tests, it must be replaced with a serviceable unit. (Use Type MD-1 capacitance bridge, R88T0941-005, to check indicator-amplifier. Use Type MD-2 tester, R88T0831-500, to check output of tank units.) The unit capacitance test, using a capacitance bridge (Simmonds Aerocessories Capacitance Bridge 387011, or equivalent), is accomplished by connecting the tester through test cables as shown in figure 6-83. Container quantity levels and the equivalent capacity readings plus tolerance factor are as follows:

QUANTITY OF LIQUID OXYGEN (LITERS)	CAPACITANCE READING (MMF)	CAPACITANCE TOLERANCE LIMITS (MMF)
0	63.5	63.0 to 64.0
1	69.3	68.7 to 69.9
2	75.1	74.3 to 75.7
3	80.9	80.0 to 81.6
4	86.7	85.6 to 87.5
5	92.5	91.4 to 93.4

The container must be placed in a level and upright position during the capacitance checks. A locally manufactured leveling platform (19 degrees) may be used to hold the container so that weld seam and mounting brackets are on a horizontal plane. To determine the quantity of liquid oxygen in the container, it will be necessary to weigh the container and to obtain quantity levels by means of container weight measurements. Since liquid oxygen weighs 40.2 ounces per liter, a container with 5 liters of liquid oxygen will weigh 201.0 ounces more than an empty container. Perform the weighing and capacitance measurements as follows:

a. Place empty container with locally manufactured leveling platform of known weight on scales.

Note

Use a weight scale with an accuracy of 0 to 400 (±0.5) ounces.

b. Connect electrical test cables between capacitance bridge (Simmonds 387011, or equivalent) and container. Measure the dry capacitance of the container. Capacitance should measure 63.5 (±0.5) mmf. Record this value.

c. Remove the capacitance bridge and weigh the empty container in ounces. Record this value.

d. Connect capacitance bridge to container. With a source of liquid oxygen available, connect the delivery hose to the filler valve on the container and fill the container with approximately 5 liters of liquid oxygen. Disconnect delivery hose from filler valve and record the actual value of this capacitance (approximately 92.5 mmf).

e. Disconnect the capacitance bridge and weigh the container and its contents in ounces. Calculate the weight of the liquid oxygen by subtracting the weight of the container (empty) from the weight of the filled container. Record this value.

f. Repeat this procedure four more times in approximately 1-liter increments (4-, 3-, 2- and 1-liter values). This may be accomplished by bleeding off approximately 1 liter of liquid oxygen each time. Bleed off each liter of liquid oxygen by connecting a delivery hose to filler valve and applying gaseous oxygen pressure to the vent port of the container.

Note

It may be easier to have the capacitance bridge connected to the container when bleeding off the liquid oxygen. By observing the decrease in capacitance bridge setting, it will be known when approximately 1 liter has been removed from the container.

g. After recording values of capacitance bridge setting and the weight in ounces for each liter of liquid oxygen, reference must be made to the chart in figure 6-84. The values of weight and capacitance must be within the limits shown. The middle line is the ideal curve of weight versus capacitance and the two outside lines are the upper and lower tolerance curve of limits. If the results of the test do not comply with the requirements of the chart, the container must be rejected.

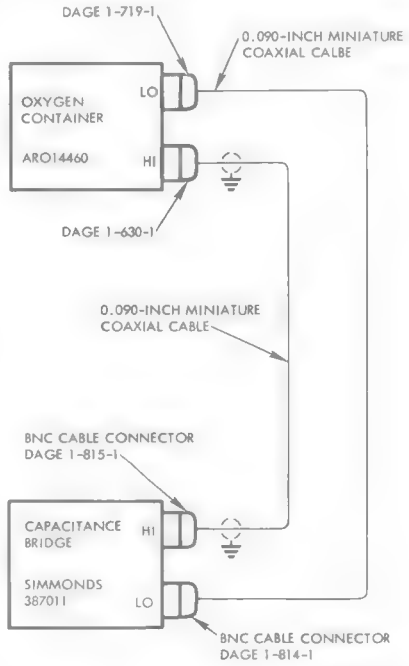


Figure No. 6-83. Adapters and Test Cables Used for Checking Oxygen Container

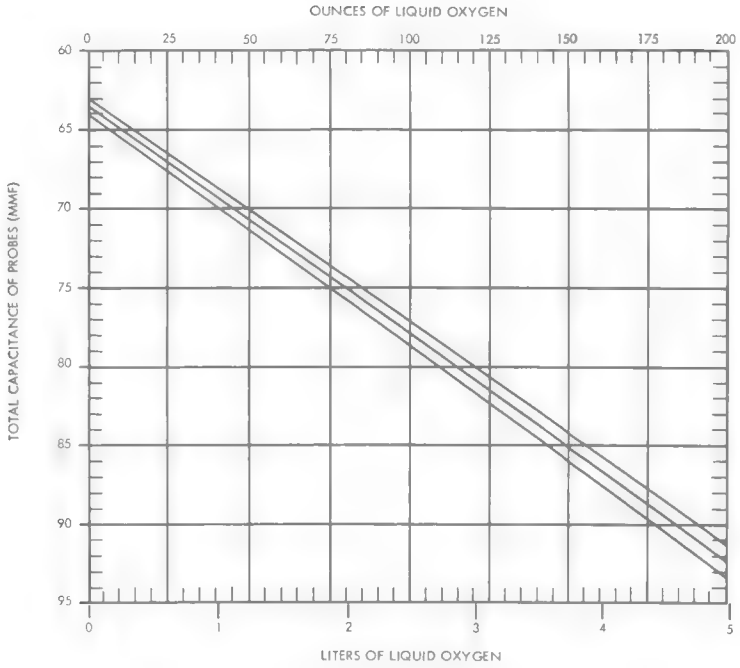


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Figure No. 6-84. Weight Versus Capacitance Chart

ALPHABETICAL INDEX

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